

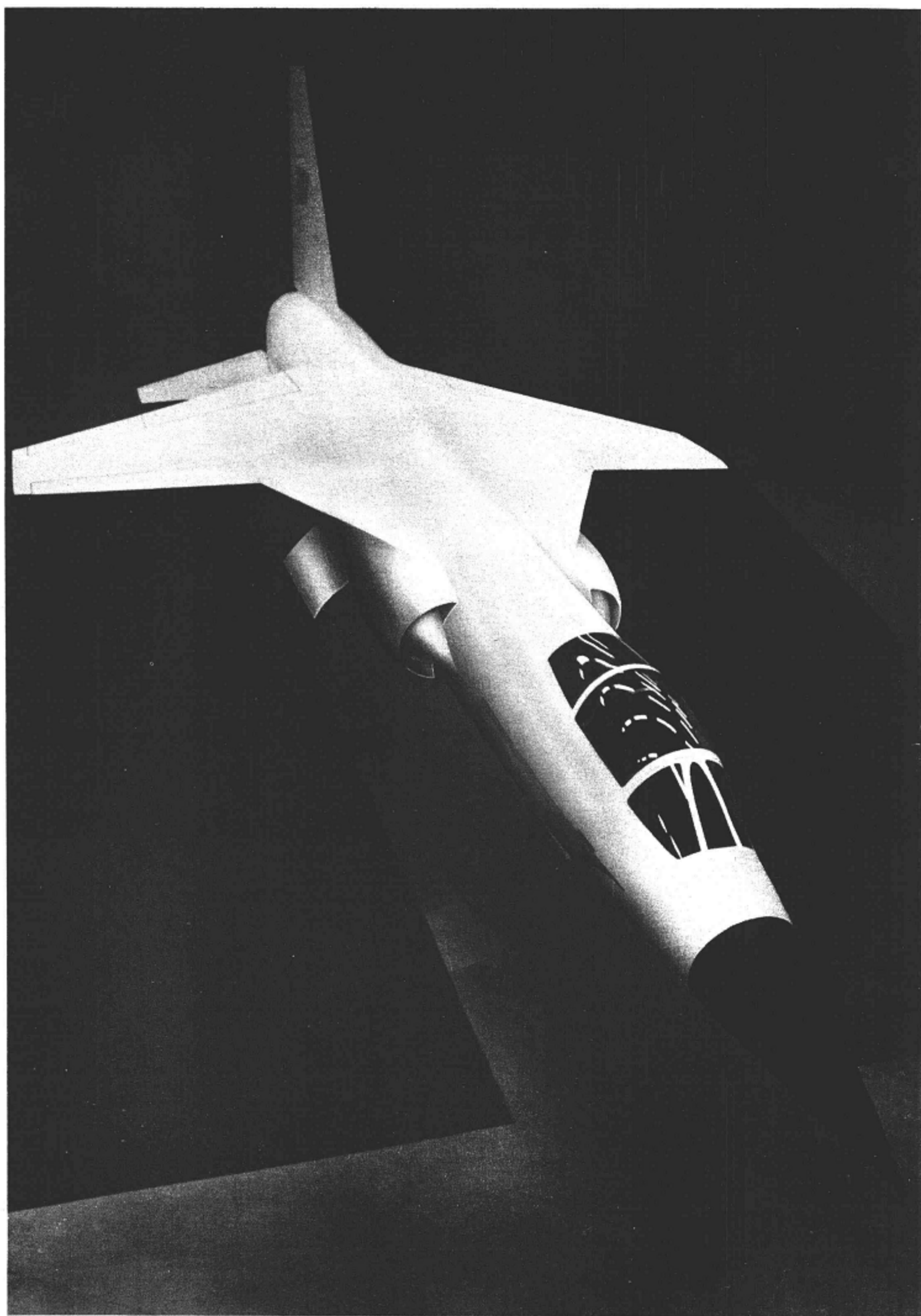


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SECTION 1 INTRODUCTION AND GENERAL DESCRIPTION

1.1 INTRODUCTION

This volume of Lockheed's proposal to the Turkish Air Force Foundation to establish an aircraft industry in Turkey covers the advanced fighter which will replace the F-104S. The advanced fighter is the Lancer. (In its earlier development phases, the Lancer was designated the CL-1200 and CL-1400.) In this volume, the Lancer weapon system program for Turkey is described in a similar manner and in the same sequence as the F-104S program proposal covered in the preceding volume.

Since the Lancer is not yet an operational weapon system, some of the production and operational data are not yet available and are estimated for the overall Lancer weapon system. However, because of the unique design and development approach to the Lancer, most of the functional elements of this advanced fighter are today in existence and in many cases operational.

One example of such a major item of Lancer equipment is the engine. The Lancer engine is operational today in USAF F-111F tactical fighters. Thrust, fuel specifics, weight, maintenance, reliability and cost data presented here for the engine are fact. The only estimates applicable to the engine are those affected by the passage of time between now and when the Turkish Industry and Air Force implement their Lancer program. Even these estimates are based on facts developed from historical USAF improvement trends for their first line operational systems and equipment.

Besides using existing functional systems, the Lancer is designed to use a significant percentage of the F-104S airframe and systems and essentially all of its external stores, both tanks and ordnance. This has been done wherever existing airframe or system components are satisfactory operationally and their use imposes no significant penalty on the Lancer mission and performance design objectives. An example is the use of the F-104 gun installation and many other equally important common systems.

Assumptions for this Proposal

The authors of this volume on the Lancer are the advanced systems development staff reporting to Lockheed's Mr. C. L. (Kelly) Johnson in his Advanced Development Projects (ADP) organization. This organization is famous for its achievements in developing and producing truly advanced weapon systems from concept to operation in extremely short time spans and for a fraction of the normal system costs.

To define a 1980 time period fighter to be produced in Turkey, starting in 1977, five basic assumptions were made. These assumptions form the basis of all the Lancer data presented in this volume.

1. The Lancer is an operational fighter in one or more air forces in 1977.
2. The Lancer is in series production in at least one factory in 1977 and will continue to be produced into the 1980's.
3. Lancer production is phased into the Turkish Aircraft Factory to sustain an even workload as the plant becomes more efficient and as the F-104S production phases down.
4. The first production version of the Lancer is defined as currently proposed.
5. The Lancer is introduced into Turkish Air Force operational squadrons replacing retired F-104 aircraft.

In each of the above assumptions, there is both justification and logic to support the assumption as stated.

In support of the first two assumptions, Lockheed and Aeritalia have reached agreement on a program for the development of the Lancer. This program provides for fabrication and flight of a Lancer prototype in the first half of 1974. Our plan provides for both Aeritalia and Lockheed to produce, coproduce, and/or license the production of the Lancer for and/or in specified countries. This first phase of development is entered into by both parties as a joint venture, after extensive analysis of all aspects of the competitive environment, military requirements, market potential, development costs, and financial soundness of the investment. The first phase will be to extend the design work already done by Lockheed on through prototype testing of the Lancer. With these test results, and the availability of a prototype to enter into "fly-off" competitions in a large number of countries facing fighter replacement requirements in the next few years, Lockheed and Aeritalia have agreed to jointly undertake an aggressive marketing plan to ensure recovery of the investment.

Of particular interest to the Turkish government in this proposal is the fact that the Lockheed/Aeritalia plan provides for licensed production of the Lancer as a follow-on to the F-104S in Turkey. The relationships and timing of Italian and Turkish production of the Lancer are illustrated in Figure 1-1.

A demonstratable prototype is the first and most important step leading to full production status of the Lancer. The time phasing of the F-104S program in Turkey and the Lancer program provides a comfortable span before a decision is required to provide a continuous production program for Turkish Industry. Lockheed's exploratory discussions with many of the present F-104 operators, and with other countries that have more imminent fighter replacement requirements, give complete confidence that this follow-on model will be in operation elsewhere well before a firm decision is necessary by the Turkish Government.

Selection of the Lancer by the Turkish government will provide the same program advantage as in Italy, where the Lancer can be phased into an existing F-104S production line. Lockheed has developed, on this basis, all of the program cost elements in its Lancer proposal to Aeritalia. This same data has been modified to accommodate supply of certain Lancer peculiar components and tooling to the Turkish F-104S factory, as a basis for the proposed Lancer production plan and cost estimates included in this volume. Therefore, the reasons for assumption 3 are simply to minimize costs and maintain a stable work force as the Turkish aircraft factory transitions from the F-104S to the Lancer. The manpower levels for this transition are illustrated in Figure 1-2.

Assumption 4 is necessary to define a Lancer configuration, to provide a firm basis for this volume. Obviously, this assumption is flexible in terms of what would actually be produced in Turkey for two reasons. First, the development, production and service implementation process of the Lockheed/Aeritalia program will undoubtedly result in some configuration improvements to the Lancer. Second, the Turkish Air Force does not need to define their precise configuration requirements before the first quarter of 1977. By then there will certainly be a number of options for final equipment selection, weapons and even specialized mission versions of the Lancer.

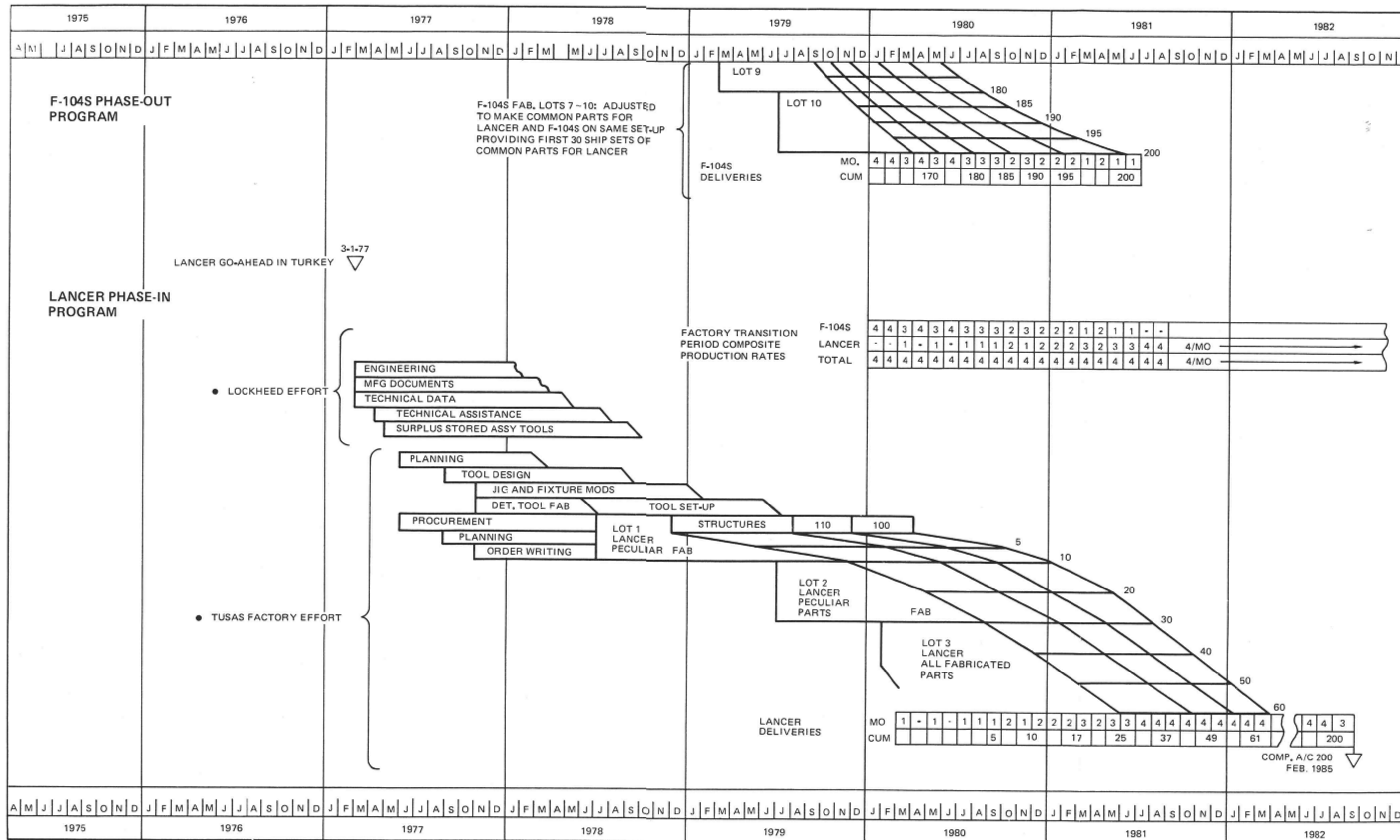


Figure 1-1. Program Plan - Factory Transition - F-104S to Lancer Production

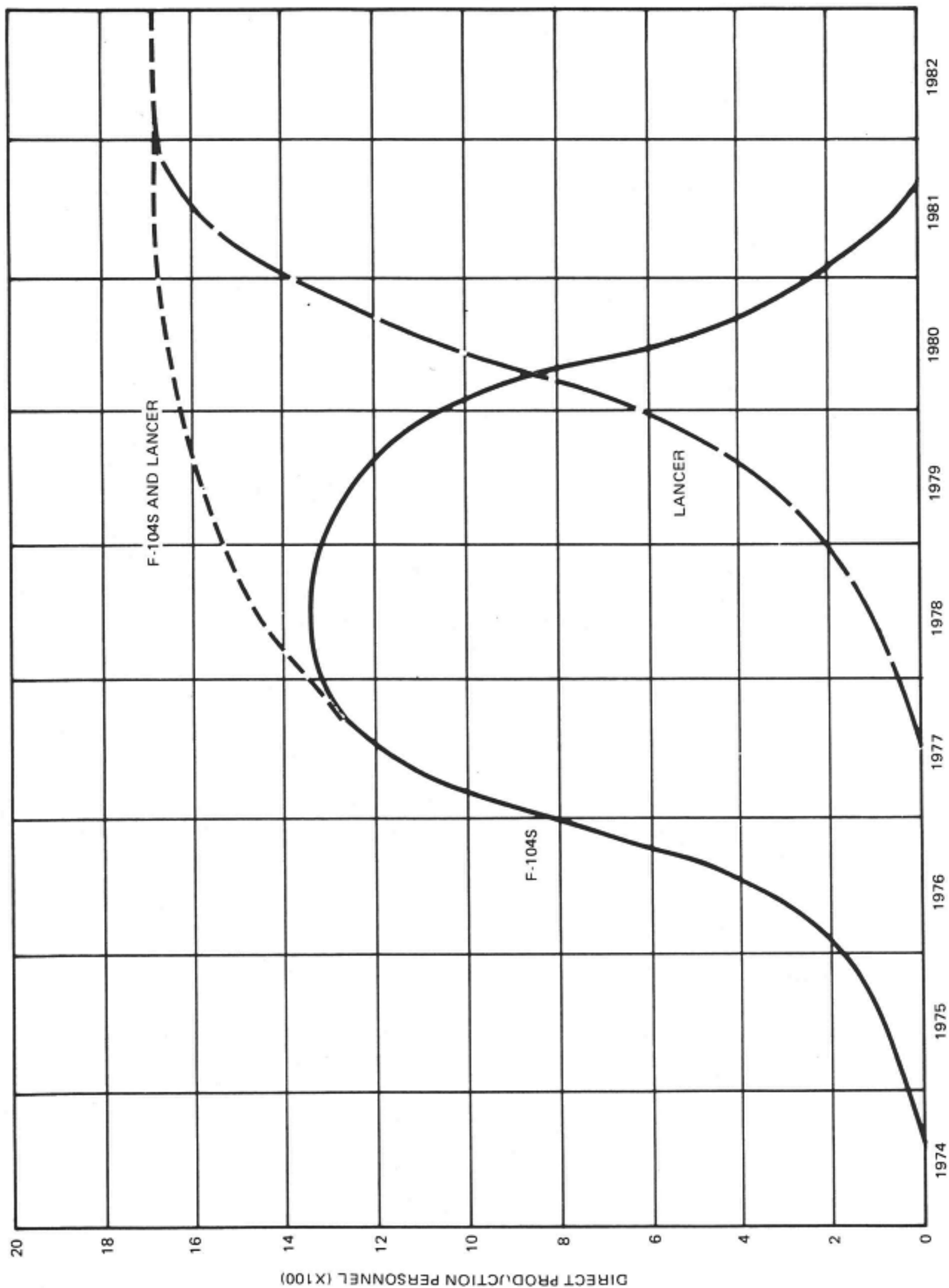


Figure 1-2. Composite Production - Parametric Manload Analysis - F-104S and Lancer

The fifth assumption is taken directly from the Turkish Air Force Foundation Volume 6 description in the final proposal request. In the case of the Lancer, this assumption ensures the minimum transition costs because so much of the squadron and base assets supporting the F-104S are retained for the Lancer.

Lancer Design Criteria

In the fall of 1969, Lockheed's Kelly Johnson surveyed the free world's fighter needs and the aircraft programs then in the planning stages. It was readily apparent that the real need for a new fighter was for greatly increased and versatile performance but at minimum cost. It is Kelly Johnson's and Lockheed's view that large numbers of new fighters are required, but they cannot be so costly that realistic national budgets force an overall reduction in the numbers of all types of weapon systems in any country's forces. Under this overall concept of maximum fighter performance balanced against minimum cost, the Lancer was conceived and developed.

Kelly Johnson translated this objective into a set of design "laws" for the Lancer, stated to his principal engineers on November 5, 1969. They are:

1. The risk and, therefore, the cost of demonstrating the performance and mission objectives shall be held to an absolute minimum.

This was to force examination and use of all existing and service proven systems, equipment and design principles from the F-104 and other Lockheed and ADP airplanes. For example, Kelly Johnson directed that the F-104's external tanks, pylons and ordnance racks be used "as is" to eliminate the cost of developing and manufacturing new designs.

2. The cost of owning the Lancer, which is the sum of design, development, tooling, production and logistics shall be the design decision making criteria against which all alternatives must be evaluated. This cost shall be held to an absolute minimum consistent with the military capability objectives.
3. Internal fuel volume and volume available for installation of specialized mission equipment must be sufficient to allow economical growth and full mission spectrum utilization of the Lancer over its operational life.

This was to preclude over emphasizing some arbitrary "designing" mission profile or performance objective that would severely limit or even preclude the Lancer from performing other mission roles.

4. The first Lancer to fly must be able to fight on at least even terms against any flying fighter in the world, regardless of the latter's cost. The ability to later improve the Lancer's effectiveness so it remains competitive with newer designs must be accommodated in the original aerodynamic configuration.



5. The primary design mission shall be to successfully engage the enemy's defensive fighters in their own airspace and then return.

This mission was selected because it is most demanding, not because it is necessarily the most important mission in any air force. In Lockheed's experience any fighter that can do this mission can do any other fighter mission with reasonably good effectiveness and efficiency. This is certainly not true of fighters designed to optimize only the attack, intercept or ground support missions.

6. From the start, the Lancer will be designed for production and operational use and the prototype will reflect these considerations. For example, the Lancer, on its first flight, will fire the gun and must be able to deliver other ordnance.
7. And finally, the Lancer must achieve stable flight for target tracking at extreme positive angles of attack.

1.2 LANCER GENERAL DESCRIPTION

The Lancer was conceived under Kelly Johnson's design laws, and under these laws the present configuration has been developed. The general arrangement is illustrated in Figure 1-3. and detailed descriptions are given in Section 4 of this volume. The Lancer design philosophy has produced readily apparent similarities to and differences from the Lockheed F-104. The similarities result from the law to use existing and service-proven systems, equipment and principles to minimize cost and risk. The differences result from new aerodynamic features which take advantage of the greatly increased thrust of the advanced technology engine and result in significant performance gains, while retaining the proven, superior features of the Starfighter, in adherence to the law that the first Lancer to fly must be superior to or fight on even terms with any flying fighter in the world.

The most obvious similarity is the forebody and cockpit - identical in size and shape to the F-104. The Starfighter's cockpit was one of the first to be designed for low drag and Mach 2+ flight, and it has received worldwide fighter pilot acceptance for comfort and excellent visibility. The forebody contains a large percentage of the proven systems and fixed equipment of the F-104, including the fully-developed M-61 gun installation. Another noticeable similarity is the outer portion of the Lancer wing. It is identical in planform to that of the F-104 wing. The fuselage side-mounted inlets are also similar to the F-104, but the centerbody "spikes" of the Lancer inlets translate automatically for optimum recovery and engine performance at all Mach numbers. They are also made larger and stronger to take advantage of the 800 knot EAS speed limit of the new engine (Mach number 1.21 at sea level). A less obvious similarity is the retention of the landing gear



design of the F-104S, the Starfighter model with the highest takeoff gross weight. A very obvious similarity is the retention of all of the F-104's external tanks and ordnance and the associated pylons, racks, adapters and launchers.

In its selection of the F-104 forebody, cockpit, items of fixed equipment, gun installation, outer wing, types of inlets, landing gear design and external stores, and in the conventional structural design of its new portions, the Lancer reduces development and qualification risks and costs. It also makes full use of a high percentage of the investment in materials, processes, tooling, manufacturing facilities, ground support equipment, spare parts and trained personnel already in existence because of the F-104 worldwide program.

The most apparent difference between the Lancer and the F-104 - the change to a high-wing, low-tail configuration - was to improve stability for target tracking at extreme angles of attack. The stability of the new Lancer configuration was verified in the first wind tunnel test to an angle of attack of 30 degrees.

Other obvious configuration differences from the F-104 are the larger wing with over 50 percent more area and the "delta" - shaped surfaces extending from the inlets to the wing leading edge. The larger wing of 311 square feet, possible with the increased thrust engine, along with automatic maneuvering flaps substantially increases maneuvering performance. The deltas decrease supersonic trim drag and increase maximum lift at all Mach numbers and in takeoff and landing. The excellent airfield performance of the Lancer, which will result in increased safety and tire and brake life, is achieved without the use of Boundary Layer Control.

A more subtle difference is increased fuselage volume for new and additional avionics or fuel, provided by adding a 30-inch barrel-section aft of the forebody retained from the F-104. The Lancer has the space and air conditioning capacity to accommodate a great variety of newly developed avionics equipment to meet individual users' needs and desires. The internal fuel capacity of the Lancer is 9075 pounds in two integral internal tanks, and in the inboard sections of the new larger wing.

New mid- and aft-body sections complete the high-wing, low-tail configuration of the Lancer. To accommodate the new Pratt & Whitney TF30-P-100 engine, the Lancer has a slightly larger aft fuselage and the overall length is increased by 30 inches to 57.25 feet.



The speedbrakes of the Lancer were not located on the sides of the aft fuselage, like the F-104, since this location would disturb the airflow over the horizontal tail located low on the aft fuselage. Wind tunnel tests of the speedbrakes located in the base of the vertical stabilizer showed good effectiveness and no adverse characteristics, and this location was adopted. The drag chute and tail hook provisions are similar to the F-104S.

The empty weight of the Lancer is 16,491 pounds and the current design maximum takeoff gross weight is 32,500 pounds. The airframe is capable of higher weights in "growth" versions with later engines of increased thrust and a strengthened landing gear.

The Lancer evolved from three years of intensive wind tunnel tests, design effort and studies and analyses of performance, stability, control, propulsion and structural methods. The continuous design evolution of the Lancer since 1969 has resulted in the completion of a large percentage of the detailed design of the airplane. At this time, a major portion of the shop release drawings are complete.

The wind tunnel test hours now total 2800. Performance and stability and control characteristics have been thoroughly evaluated to Mach number 2.4. Lockheed wind tunnel test results over the Mach number range from 0.6 to 1.4 have been verified by duplicate tests performed in the NASA Ames Research Center 11 by 11 foot transonic tunnel, and in the Cornell Aeronautical Laboratory 8 foot transonic tunnel.

The development of the Lancer configuration is shown by the photographs of the Lockheed wind tunnel models in Figure 1-4. From the upper left, these show the evolution of the configuration to the Lancer as it appears today in the lower right photograph.

These tunnel tests are the basis for the predicted performance presented in this proposal. That performance shows that the Lancer is truly a balanced design between the conflicting requirements of maneuvering and acceleration capability. Sufficient internal fuel is carried for exceptional fighting endurance at long ranges from base. The size and internal fuel capacity of the Lancer also provide the complementary capability of a potent ground attack fighter.

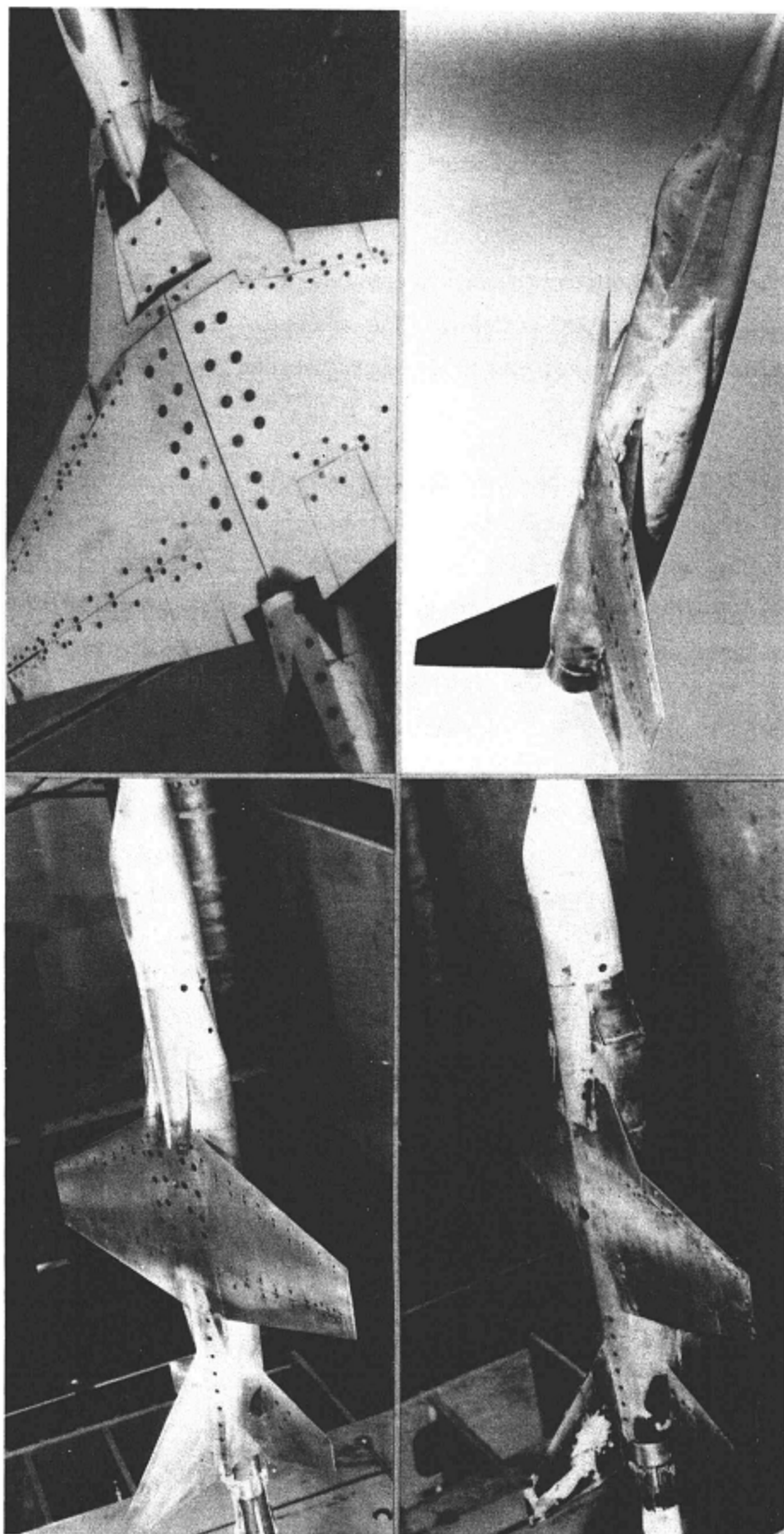


Figure 1-4. Wind Tunnel Configuration Developments

The current Lancer has a specified maximum sustained Mach number of 2.2. This sustained maximum is selected since, in Lockheed's opinion and based on the Viet Nam combat experience, any higher Mach number is not sufficiently useful to justify cost and weight increases for cockpit glass and detailed systems refinements for higher Mach numbers. The Lancer basic airframe and engine, however, are now fully capable of flight to Mach number 2.4 for short periods of time, and have been wind tunnel tested to this Mach number. If future customer desires or needs dictate, the Lancer can be modified for flight at even higher Mach numbers, in growth versions.

Alternate configurations of the Lancer are also possible, due to the concept of using the F-104 forebody. Examples are the two-place trainer and reconnaissance or special mission airplanes.

Finally, in adherence to Kelly Johnson's "law" that the future growth capability to improve the Lancer must be accommodated in the original aerodynamic configuration, the current Lancer's inlets and aft fuselage have been designed to accept the new F401-PW-400 engine, of the same Pratt & Whitney "family" as the TF30-P-100, when that engine is fully developed and operational. The F401 engine is currently being developed for the U. S. Navy's F-14B fighter. The later installation of the F401 engine in the Lancer will provide 3000 additional pounds of sea level static thrust over that of the TF30-P-100 and will further increase the Lancer's performance and load carrying ability at all speeds and altitudes.

The Lancer design is valid. That is, it uses proven systems and components in satisfying a NATO mission requirement as the way to limit development risk and program cost.

SECTION 2 PERFORMANCE

The Lockheed Lancer will have significantly increased capability over the current inventory of fighters in all phases of performance: takeoff, landing, climb, cruise, acceleration, combat maneuvering, and flexibility in all types of fighter missions. These advances in fighter performance achieved by the Lancer are the result of two basic design principles which deserve to be examined in detail:

- Maintain high excess thrust (thrust minus drag) to weight ratios at all Mach numbers and altitudes in the flight envelope. This will result in a fighter which does not sacrifice acceleration for subsonic maneuvering performance. The Lancer is superior as an interceptor at high supersonic Mach numbers and altitudes as well as in subsonic, maneuvering combat. The aircraft can also sustain high maneuvering performance at transonic and supersonic Mach numbers. This design principle is achieved by the selection of the advanced technology Pratt & Whitney TF30-P-100 engine for high thrust and by advancements in aerodynamic design for lowered drag at high lift. The flight proven TF30-P-100 engine develops 25,100 pounds of Sea Level Static Thrust, over 7000 pounds more than the G.E. J79-19 engine. This engine is combined with a translating centerbody inlet that provides high recovery and low inlet drag at all Mach numbers. The Lancer wing planform, area, and thickness ratio have been optimized in wind tunnel tests for the best "balance" between subsonic and supersonic lift-to-drag ratios. The "deltas" on the inner wing leading edge were conceived and developed to reduce supersonic trim drag at high lift. Subsonic lift-to-drag ratios are enhanced by use of the wing leading edge and trailing edge flaps for maneuvering. The schedule of flap deflections has been determined in wind tunnel tests and varies from 15 degrees leading edge and 15 degrees trailing edge at low, subsonic Mach numbers to flaps up at 0.95M and higher and supersonic Mach numbers. The flaps are scheduled automatically as a function of Mach number, dynamic pressure, and load factor when the pilot selects "Maneuver Flaps" by a switch on the control stick.

Design for a high internal fuel fraction (the ratio of internal fuel capacity to either combat weight or design maximum takeoff gross weight). When this is efficiently accomplished, endurance is provided in intercept and air-to-air combat beyond that currently achieved or even planned for many

programmed high performance fighters. High internal fuel quantity capability also provides obvious flexibility in ground attack missions by not sacrificing external ordnance for external fuel. The Lancer has an internal fuel capacity of 9075 pounds and a maximum takeoff weight of 32,500 pounds. This is an internal fuel fraction of 28 percent, at takeoff and it is 41 percent at the half-internal fuel Design Combat Weight of 22,236 pounds. In designing the Lancer for high internal fuel fraction, care was taken to not increase the fuselage diameter (and drag) beyond that dictated by the sizes of an acceptable radar antenna, the pilot, and the engine. The Lancer internal fuel volume is achieved by added length and inboard wing tanks. The added fuselage length has the additional benefit of reducing horizontal and vertical tail sizes while maintaining acceptable levels of stability and control. In Lockheed's view, there is only one disadvantage to a high internal fuel fraction achieved without increasing drag; it is traditional to compare combat performance at the arbitrary "half-internal fuel weight", and this weight is higher with higher internal fuel fraction. For this reason, a curve showing the effects of combat weight on maneuver performance is included in Section 2.10 (Figure 2-59).

Representative performance data are presented in the following sections to show the Lancer's air-to-air combat, intercept, and ground attack capability. Airfield performance is also shown. All performance is based on a 1962 U.S. Standard Day atmosphere, with additional takeoff and landing performance for an ambient temperature increase of 20°C. The 1962 U.S. Standard Day is currently in use in the United States. It is equivalent to the ICAO Standard Day to over 60,000 feet. The Lancer's performance is based on 2800 hours of wind tunnel tests culminating in the current configuration, and on the operational status of the flight-proven TF30-P-100 engine.

2.1 TAKEOFF AND LANDING

The Lancer's takeoff and landing speeds and ground distances are presented in Figures 2-1 through 2-4, for sea level and 1500 foot field elevations, on a Standard Day and for an ambient temperature increase of 20°C.

Airport performance of the Lancer is based on wind tunnel tests of the configuration with takeoff and landing flaps, including tests with a ground plane installed. Liftoff and touchdown speeds are determined at the ground clearance angle of 12.25 degrees with the main landing gear strut compressed to static extension at takeoff weight. This results in lift off at 15.5 percent above stall speed and touchdown at 10.5 percent above stall speed. The stall speed is defined by the maximum usable angle of attack of 21.5 degrees, away from the ground.

All of the Lancer takeoff performance is calculated for the forward c.g. limit of 12 percent m.a.c. All landing performance is calculated for a representative aft c.g. of 24 percent m.a.c. and assumes a three second free roll at touchdown speed before brake application and drag chute deployment.

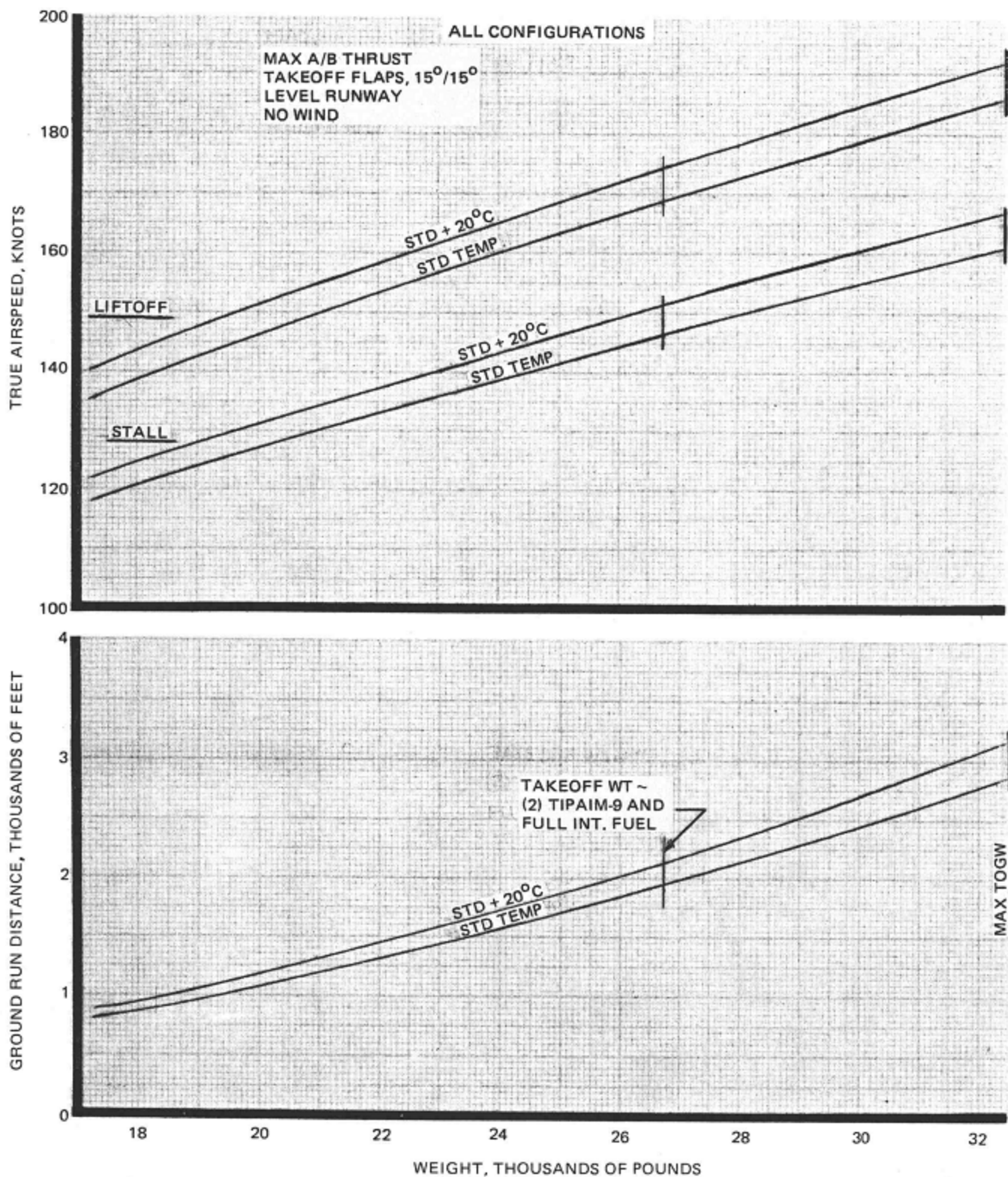


Figure 2-1. Takeoff Speed and Distance, Sea Level

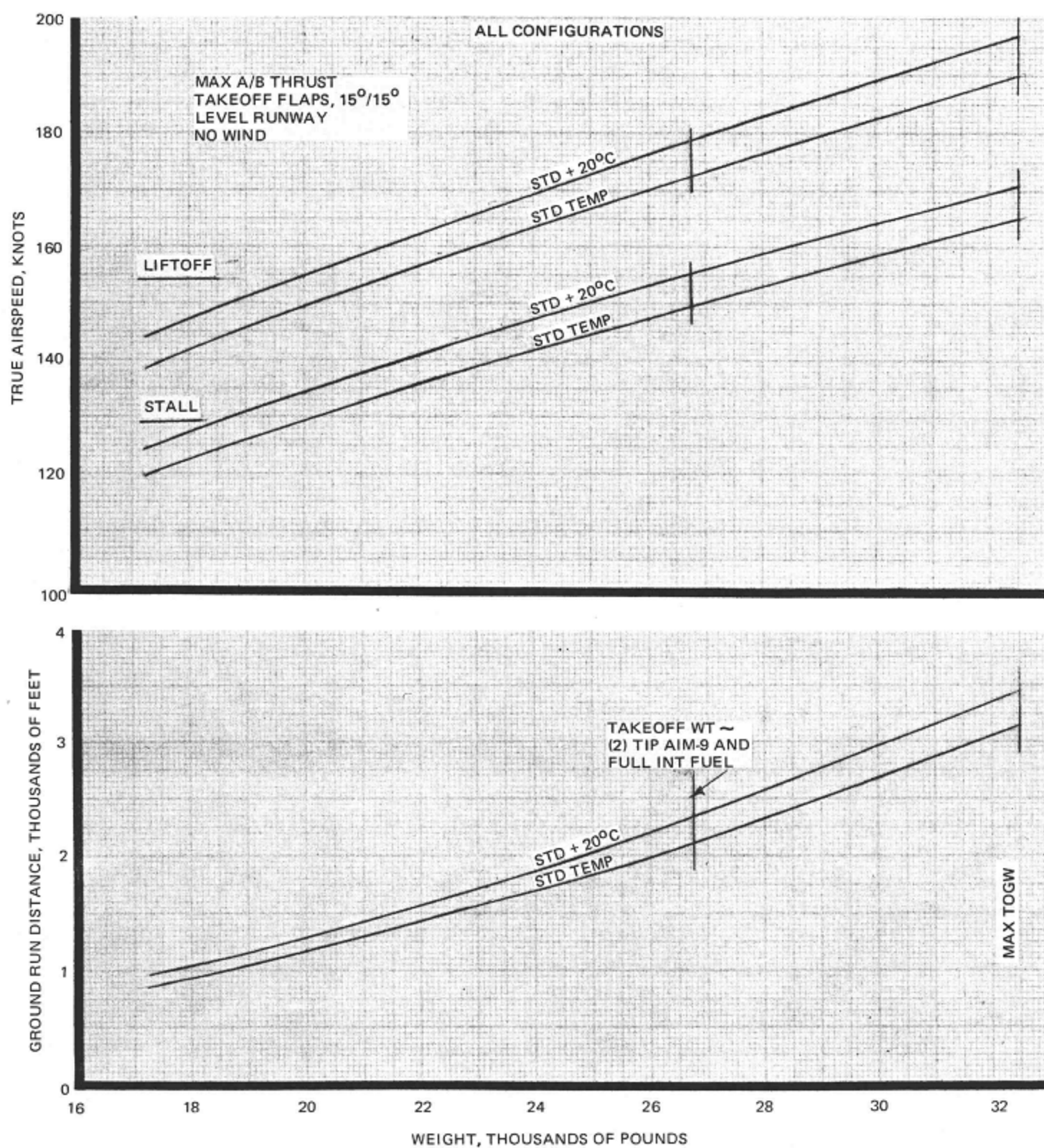


Figure 2-2. Takeoff Speed and Distance, 1500 Ft. Altitude

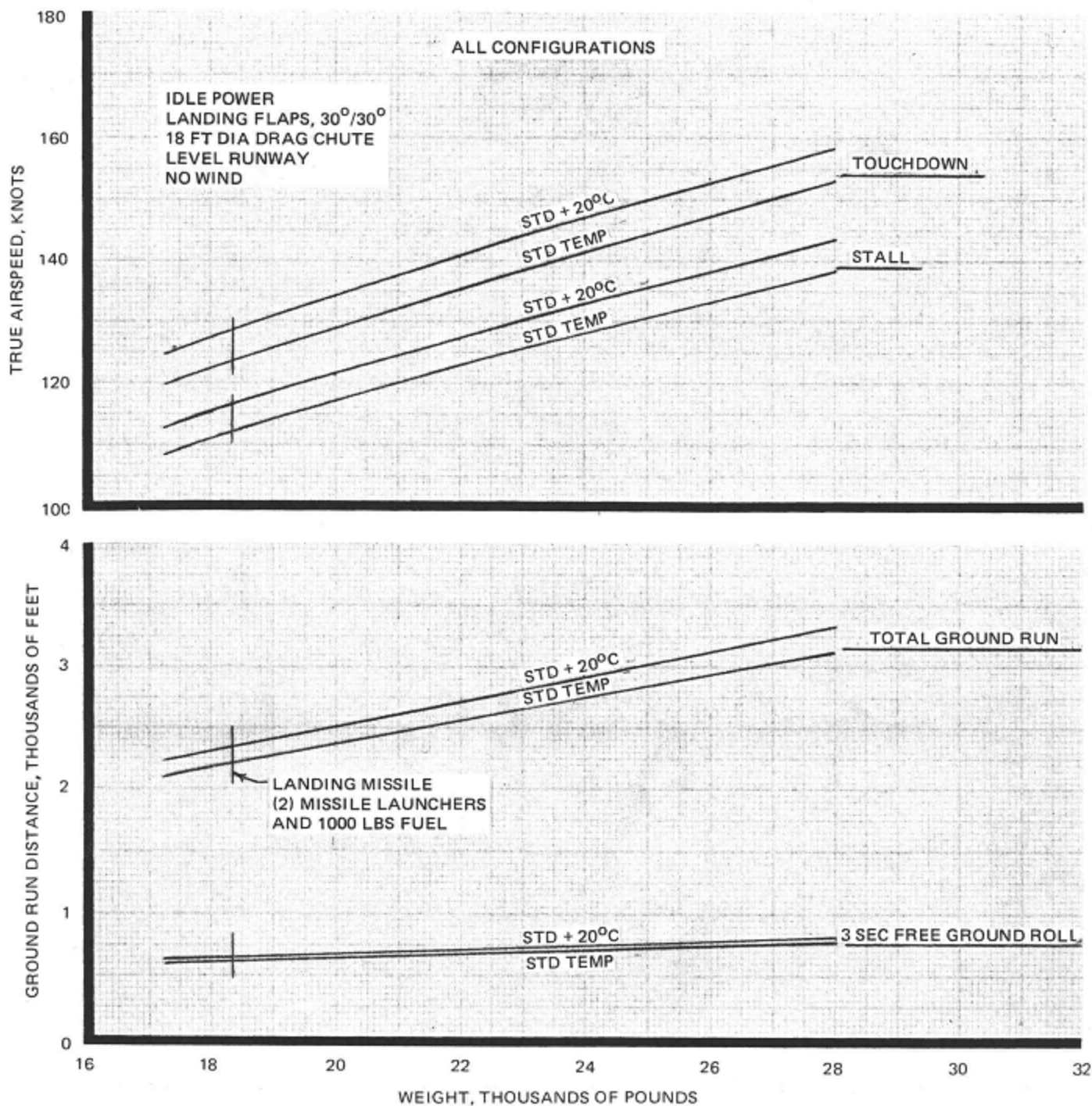


Figure 2-3. Landing Speed and Distance, Sea Level

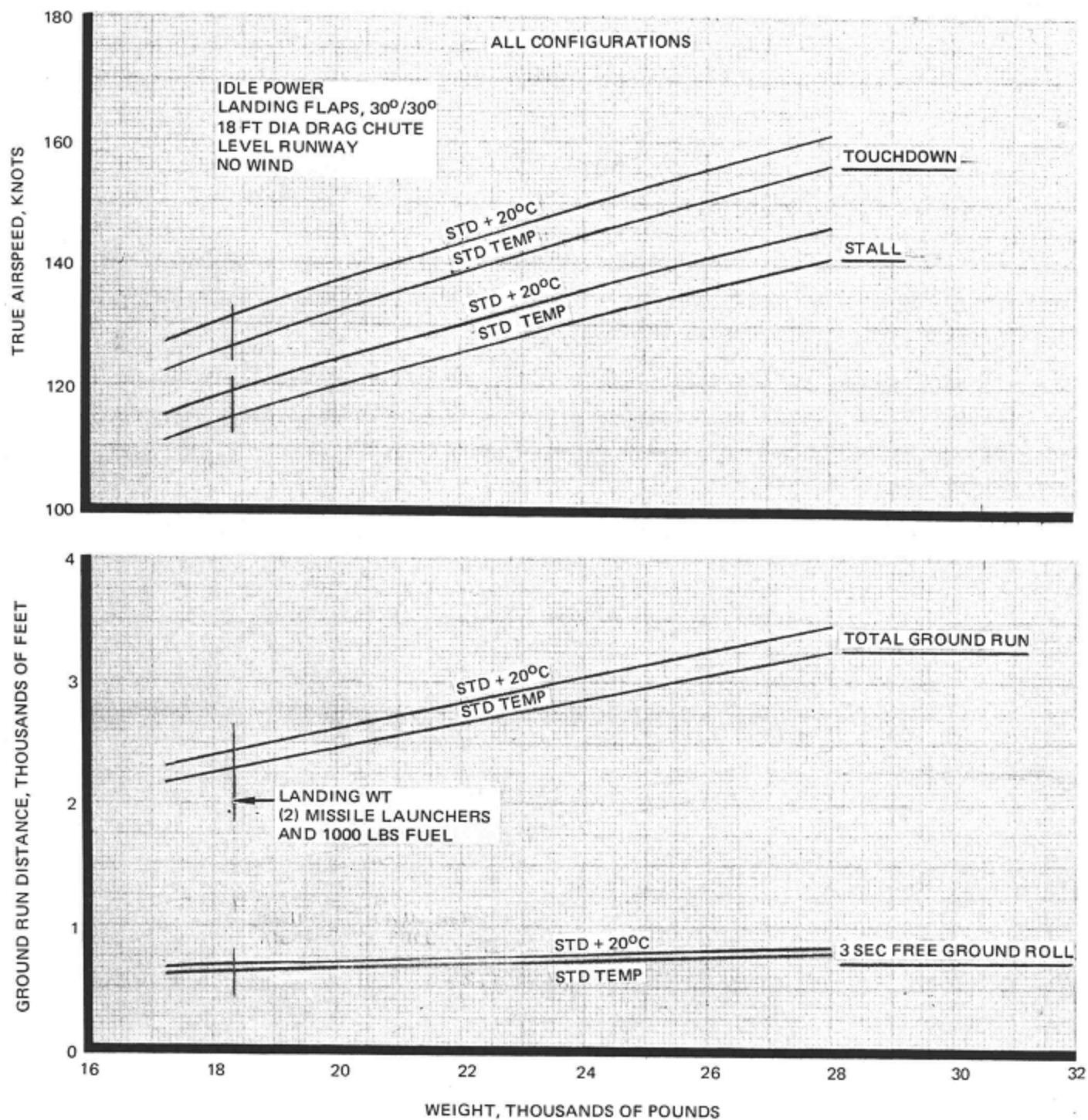


Figure 2-4. Landing Speed and Distance, 1500 Ft. Altitude

2.2 FLIGHT ENVELOPES

The level flight envelope of the Lancer with 50 percent of internal fuel is presented in Figure 2-5 for three configurations: clean; with two wing-tip-mounted AIM-7 Sparrow missiles; and with two tip-mounted AIM-9 Sidewinder missiles, for both maximum afterburning and intermediate thrust. Automatic maneuvering flaps are used where appropriate and placards and maximum usable lift limitations are indicated. The afterburner blowout boundary shown on Figure 2-5 applies to all Lancer operation, but, for expediency, is not repeated on Figures 2-22 through 2-48.

2.3 CLIMB AND ACCELERATION

Representative Lancer subsonic and supersonic climb performance is shown in Figures 2-6 through 2-8. The subsonic climb speed schedules are for minimum time to climb. Level flight accelerations from Mach number 0.9 to Mach 2.0 at 35,000 feet are presented in Figure 2-9. The climb and acceleration performance are calculated by the "weight reduced" method. Subsonic climbs are initiated with full internal fuel minus an allowance for engine start, warmup, taxi, takeoff, and acceleration to climb Mach number. Initial weights for the supersonic climbs and accelerations are with 50% of internal fuel.

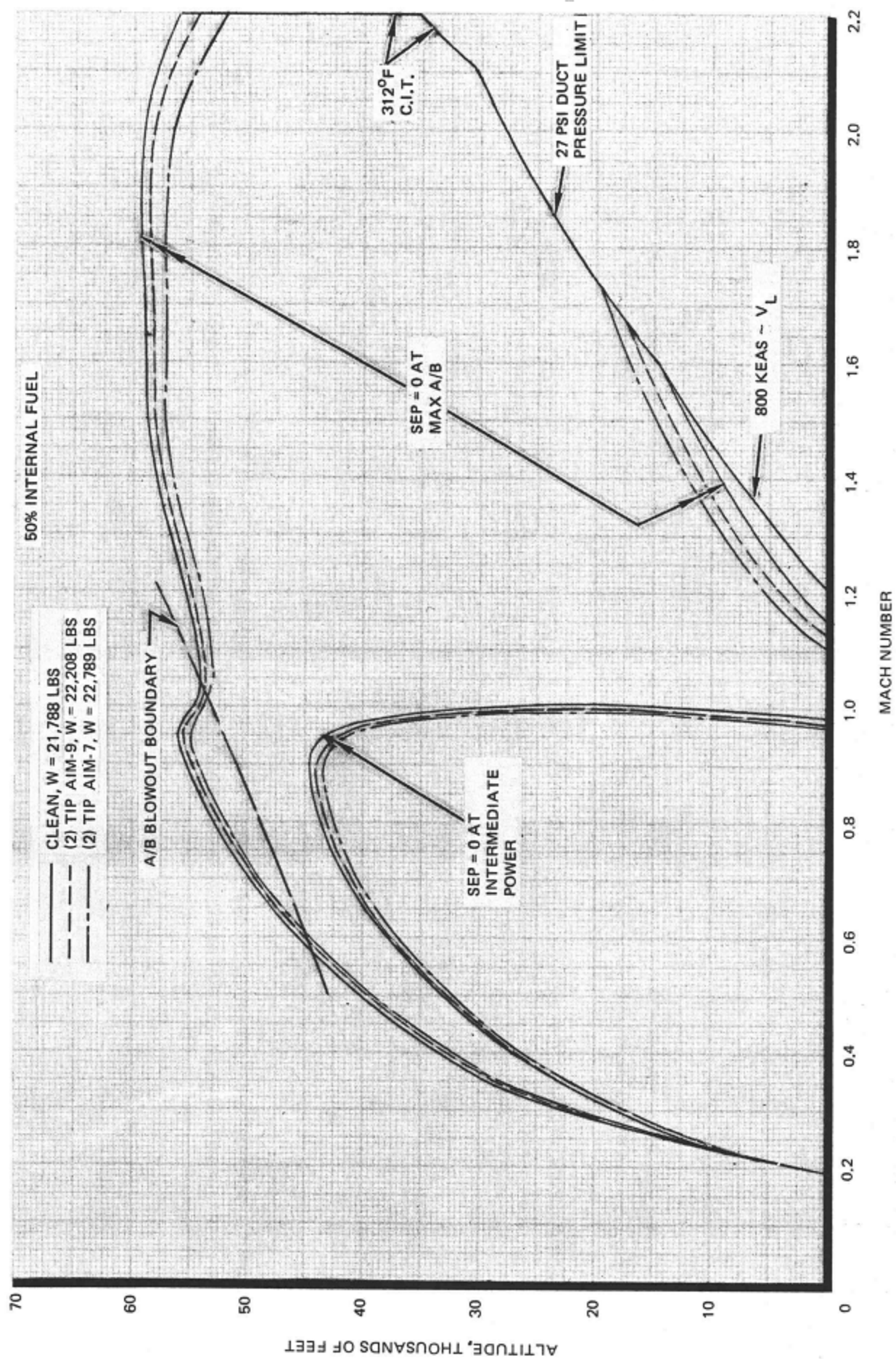
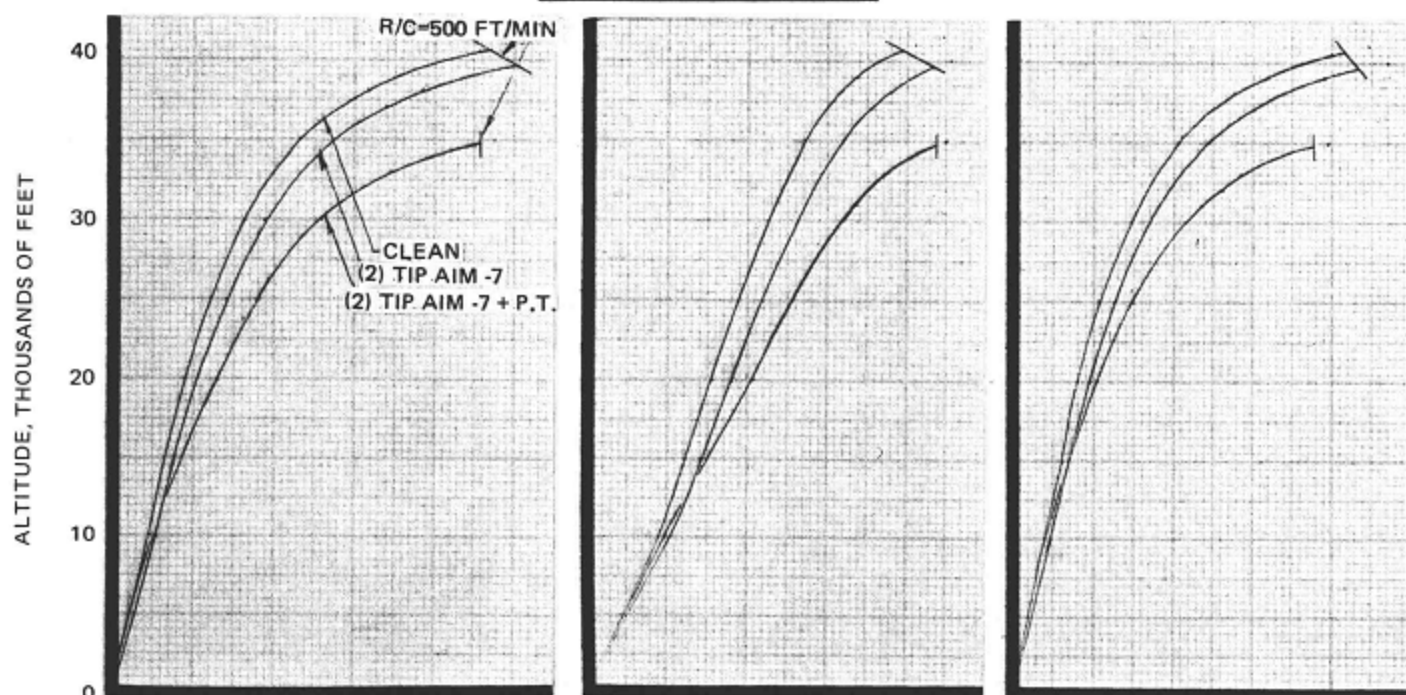


Figure 2-5. Flight Envelope

START CLIMB WITH FULL FUEL LESS TAKEOFF AND ACCEL. FUEL

ALL WEATHER INTERCEPTOR



FIGHTER - BOMBER

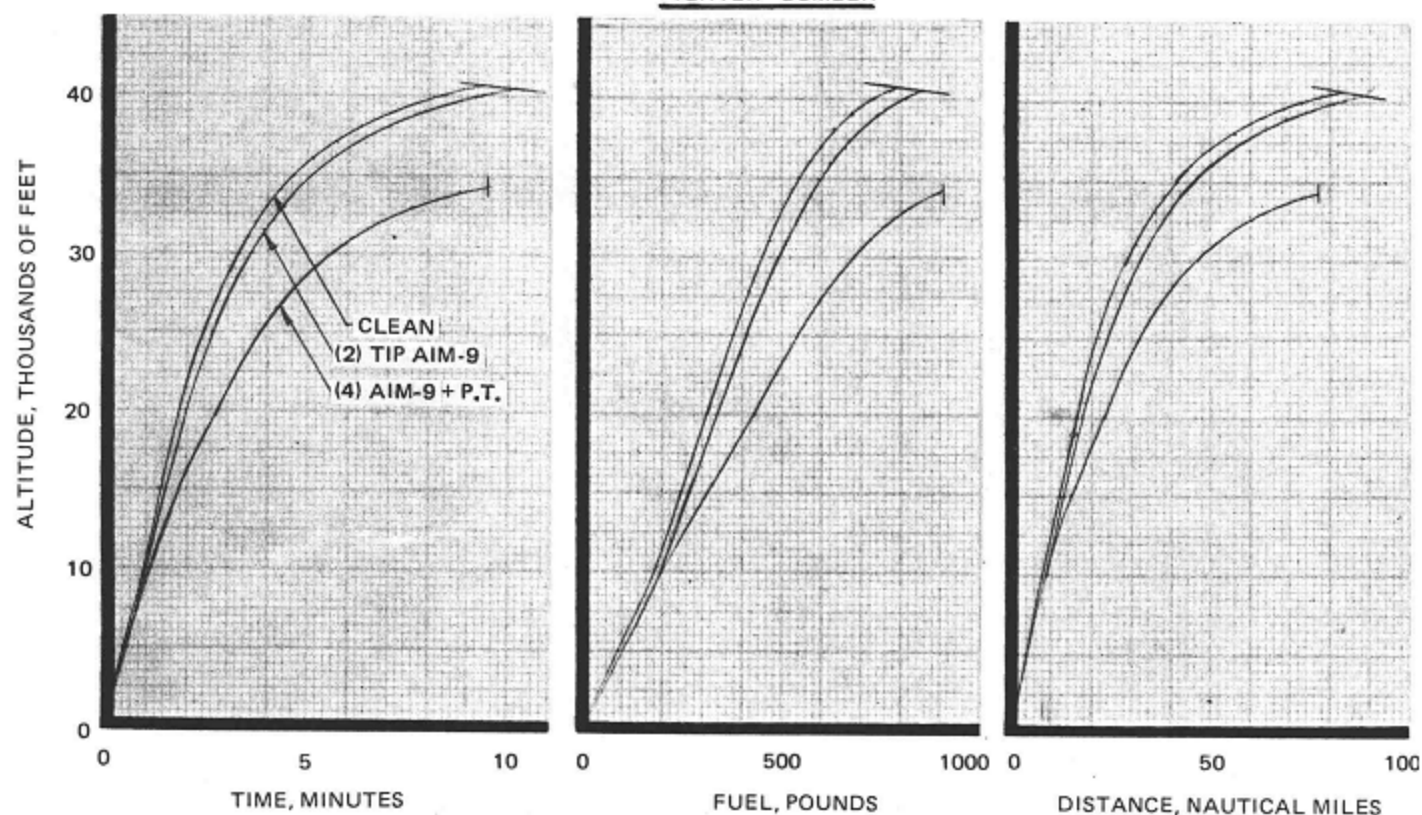


Figure 2-6. Subsonic Climb Performance: Intermediate Power

CLIMB M=0.9, START CLIMB WITH FULL FUEL LESS TAKEOFF AND ACCEL, FUEL

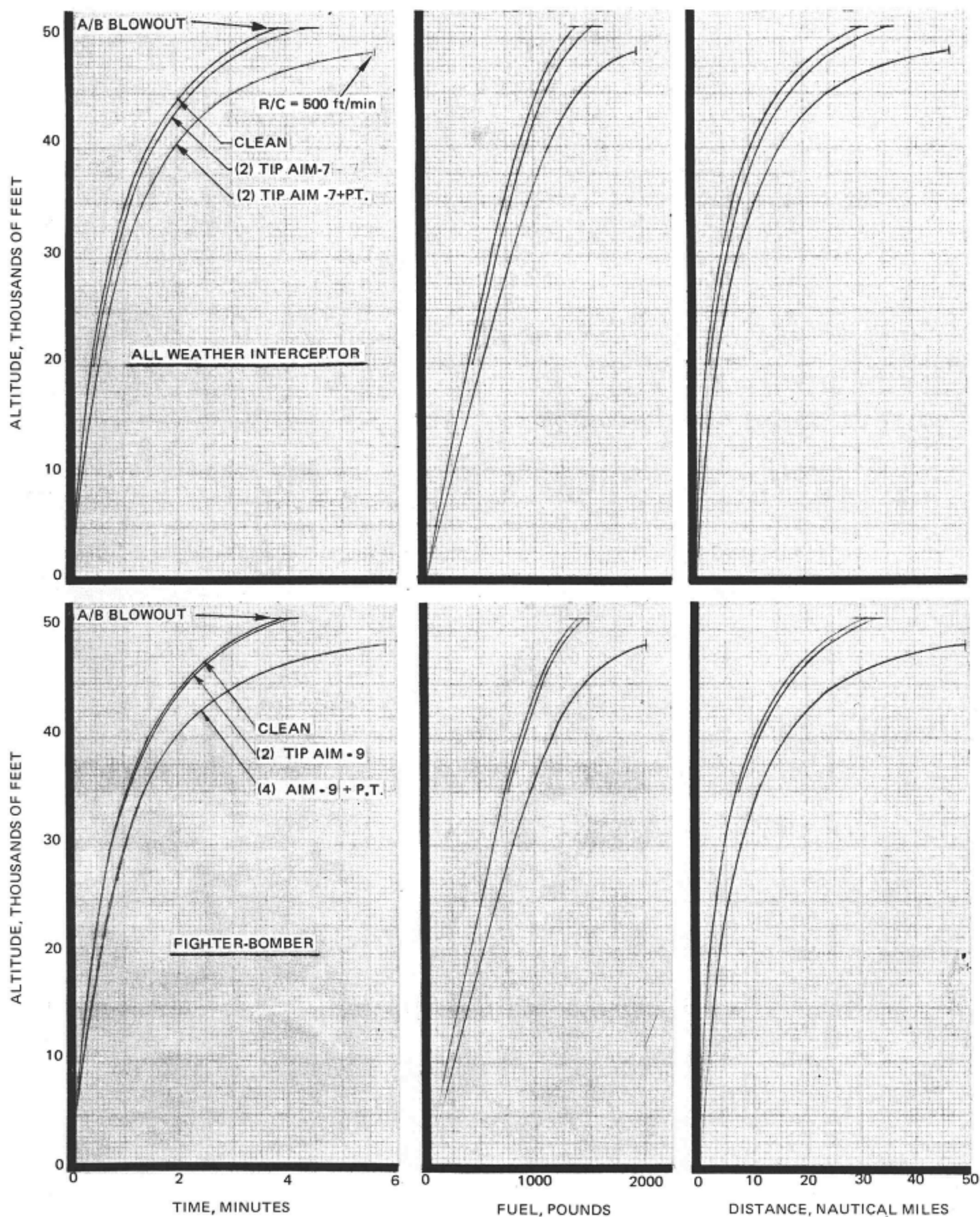
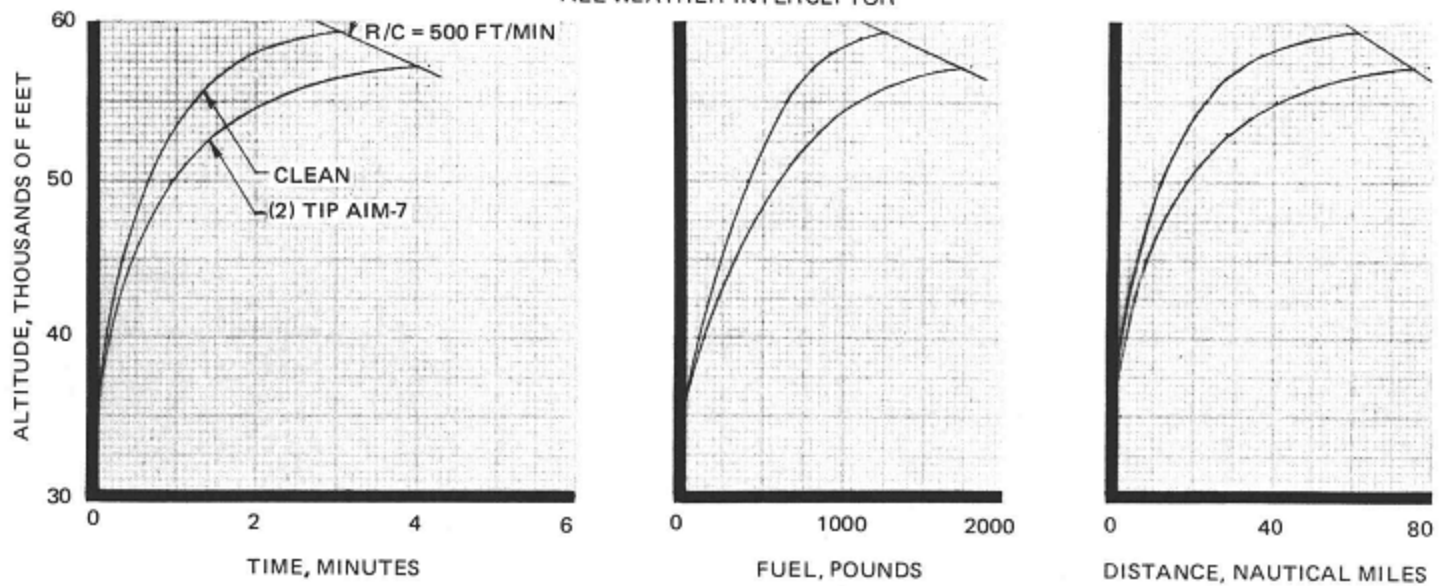


Figure 2-7. Subsonic Climb Performance: Max A/B



MAX A/B, INITIAL WEIGHT WITH 50% INTERNAL FUEL

ALL WEATHER INTERCEPTOR



FIGHTER-BOMBER

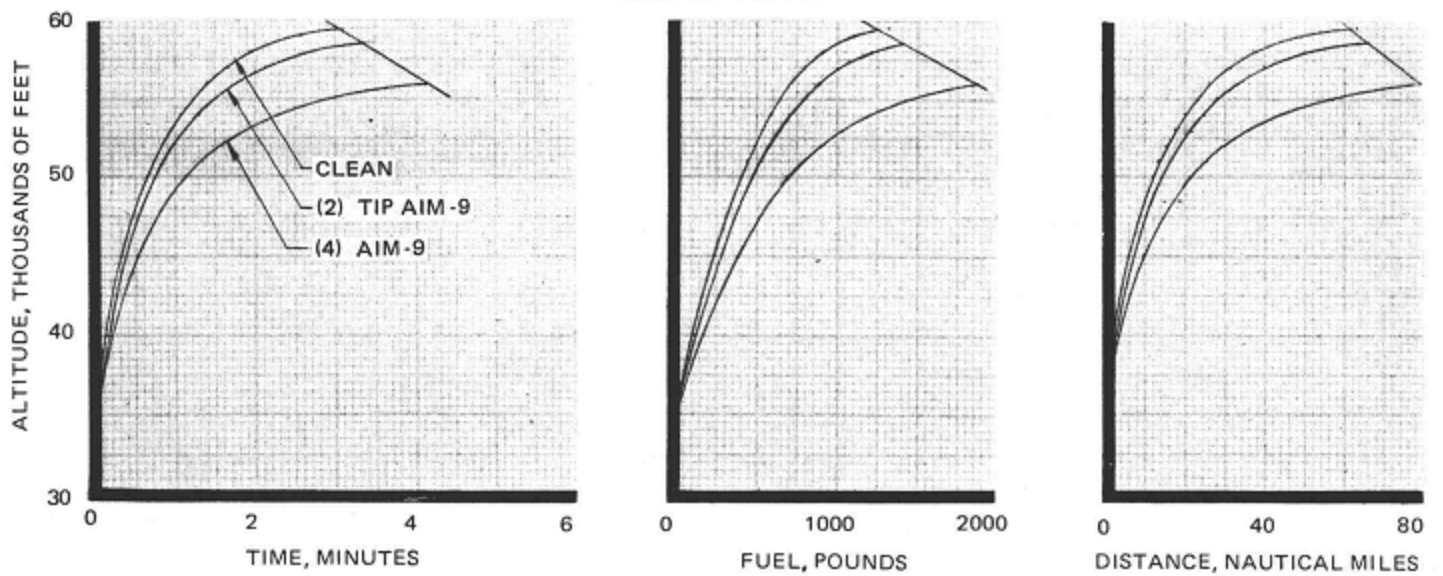


Figure 2-8. Mach 2.0 Climb Performance



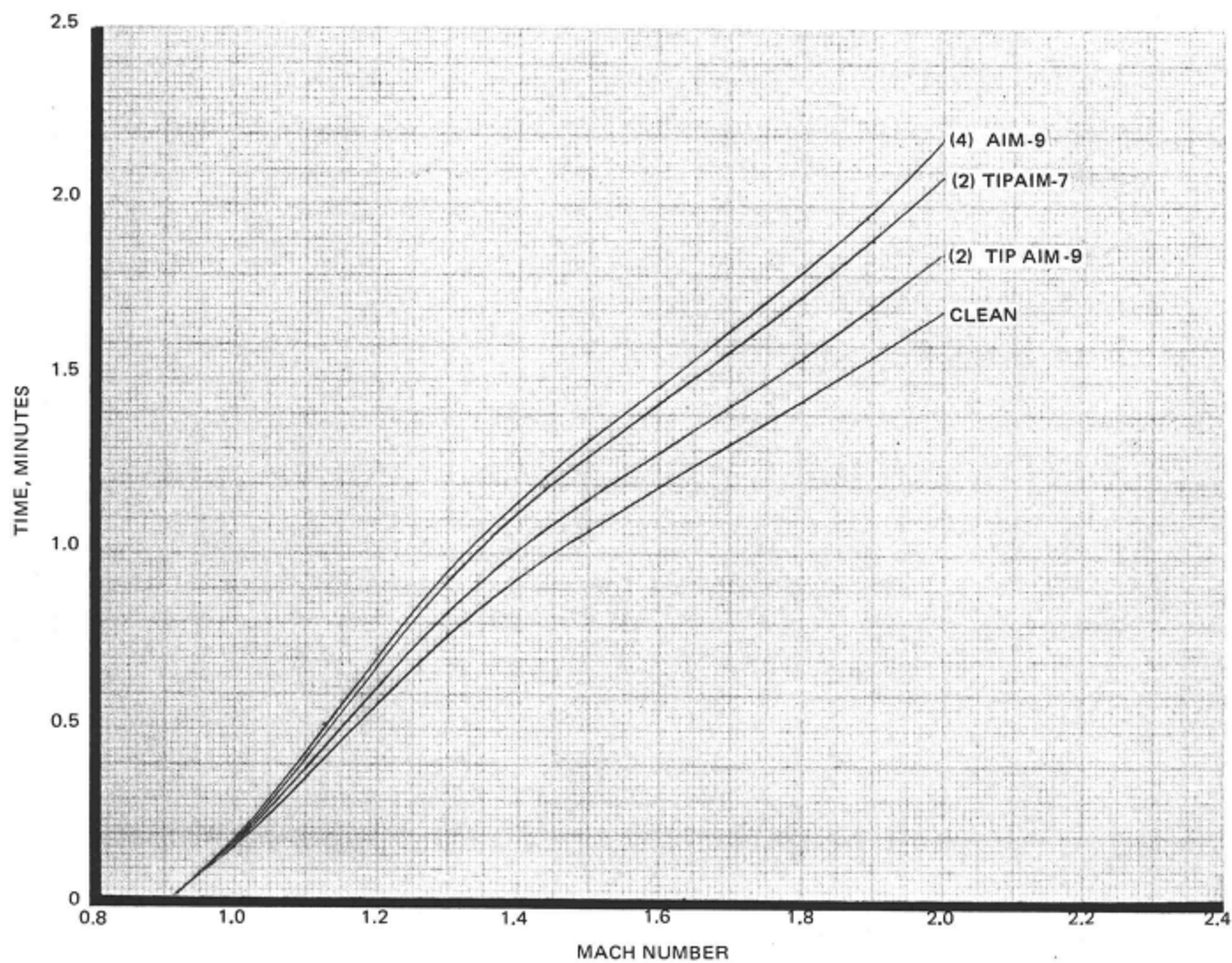


Figure 2-9. Level Flight Acceleration at 35,000 Feet

2.4 MISSIONS

The mission capabilities of the Lancer are presented in Figures 2-10 to 2-21 on a variety of profiles. These include a combat air patrol mission, two types of subsonic intercept missions and a supersonic intercept mission. Each of these is shown for an all-weather configuration with AIM-7 Sparrow missiles and for fighter bomber missions with AIM-9 Sidewinder missiles. The maximum ordnance load that can be carried versus combat radius is shown for two counterattack mission profiles, carrying 1, 3, 5, or 7 MK-83 bombs as a representative load. Ferry mission capability with wing tip and pylon-mounted external tanks is also shown. The tanks are retained throughout the ferry mission.

2.5 MISSION GROUND RULES

The Lancer mission capabilities are detailed in Figures 2-10 through 2-21. The following criteria are used in the calculation of these mission profiles:

- Fuel flows are not increased for a service allowance.
- Fuel allowance calculation for initial ground operation includes fuel for 6 minutes at an installed thrust-to-weight ratio of 0.2 for engine start, warmup, and taxi for all except the point and supersonic intercept missions. Fuel allowances for takeoff and acceleration to climb speed are calculated for all missions by the following equation:

$$\text{Fuel} = \frac{W_1 V_1}{g(T-D)_1} \frac{FF_o + FF_1}{2}$$

Where

- W = weight
- V = velocity
- T = thrust
- D = drag
- FF = fuel flow
- g = acceleration of gravity

and

- o = sea level, static conditions
- 1 = sea level, climb Mach number conditions

- JP-4 fuel is used with a heating content of 18,400 BTU per pound and a density of 6.5 pounds per U.S. gallon.
- Optimum cruise segments are computed at the faster of the two speeds which yield 99% of maximum nautical miles per pound of fuel, at any Mach number and altitude condition.
- The M-61 gun and 500 rounds of ammunition are carried on all missions.
- All missions are calculated at the appropriate airplane weights and drags for the particular configuration, including the incremental weights and drags of external stores, tanks, pylons and armament racks, as shown in Table 2-1.
- All missions include a landing reserve fuel allowance for 20 minutes optimum loiter at sea level.
- All subsonic performance assumes the use of maneuvering flaps wherever a performance gain results compared to the flaps up configuration.

TABLE 2-1.
EXTERNAL STORES WEIGHTS

Add to Clean Zero Fuel Weight For:

(2) Wing Pylons at W.S. 112 - For Pylon Tanks	275 lbs.
- With AIM-7 Launchers	414
- With AIM-9 Launchers	427
- With Bomb Rack	324
- With Bomb Rack and TER	420
(2) Wing Pylons at W.S. 152 - With AIM-7 Launchers	433
- With AIM-9 Launchers	315
- With Bomb Rack	242
(1) Fuselage Centerline Bomb Rack	127
(2) Wing Tip Missile Launchers - For AIM-7	101
- For AIM-9	110
(2) 170 Gallon Tip Tanks - Empty	460
- Full	2670
(2) 195 Gallon Pylon Tanks - Empty	356
- Full	2891
(2) AIM-7 Sparrow Missiles	900
(2) AIM-9 Sidewinder Missiles	310
(1) MK-83 LDGP Bomb	1000 lbs.

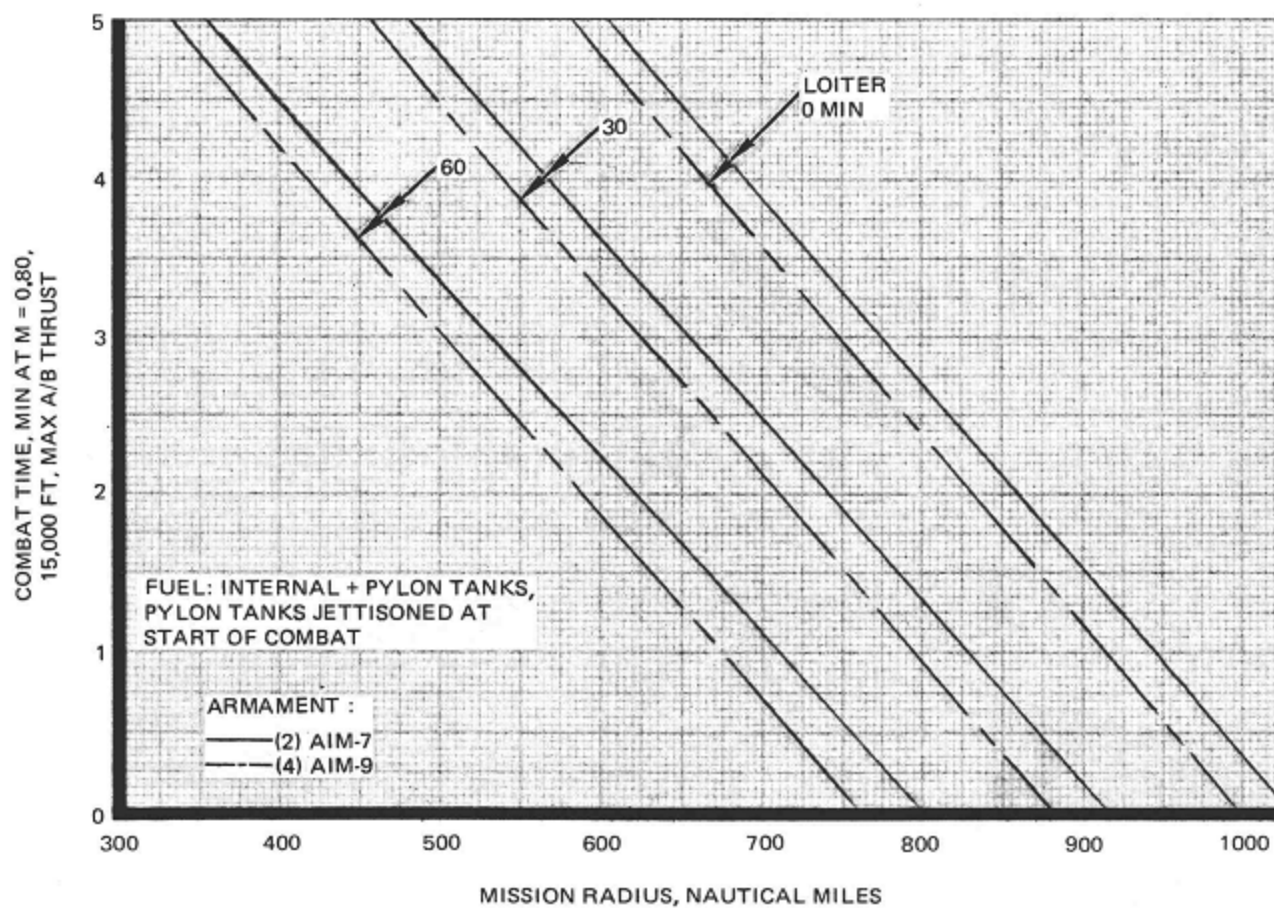
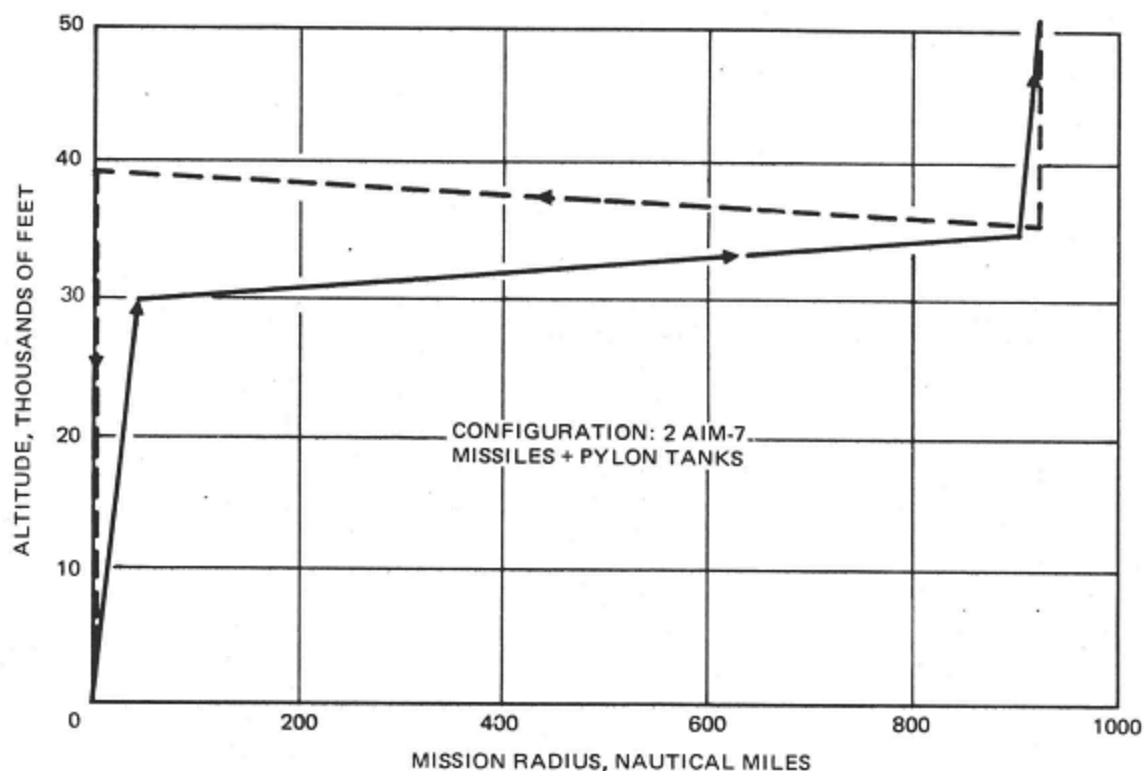


Figure 2-11. Combat Air Patrol Mission Performance



MISSION PROFILE:

1. TAKEOFF ALLOWANCE: FUEL USED FOR 6 MINUTES AT $T/W = 0.20$ + FUEL TO TAKEOFF AND ACCELERATE TO CLIMB SPEED AT MAXIMUM POWER.
2. INTERMEDIATE POWER CLIMB TO OPTIMUM CRUISE ALTITUDE.
3. OUTBOUND CRUISE, OPTIMUM MACH AND ALTITUDE. EXTERNAL FUEL TANKS JETTISONED WHEN EMPTY.
4. MAXIMUM POWER CLIMB TO 51,000 FEET.
5. COMBAT: 5 MINUTES AT MAXIMUM POWER AT 51,000 FEET, $M = 0.90$, FIRE MISSILES AND AMMO.
6. DESCEND TO OPTIMUM CRUISE ALTITUDE; NO FUEL, TIME OR DISTANCE CREDIT.
7. RETURN CRUISE, OPTIMUM MACH AND ALTITUDE.
8. DESCEND TO SEA LEVEL. NO FUEL, TIME OR DISTANCE CREDIT.
9. RESERVES: FUEL FOR 20 MINUTES SEA LEVEL LOITER.

Figure 2-12. Subsonic Area Intercept Mission Profile

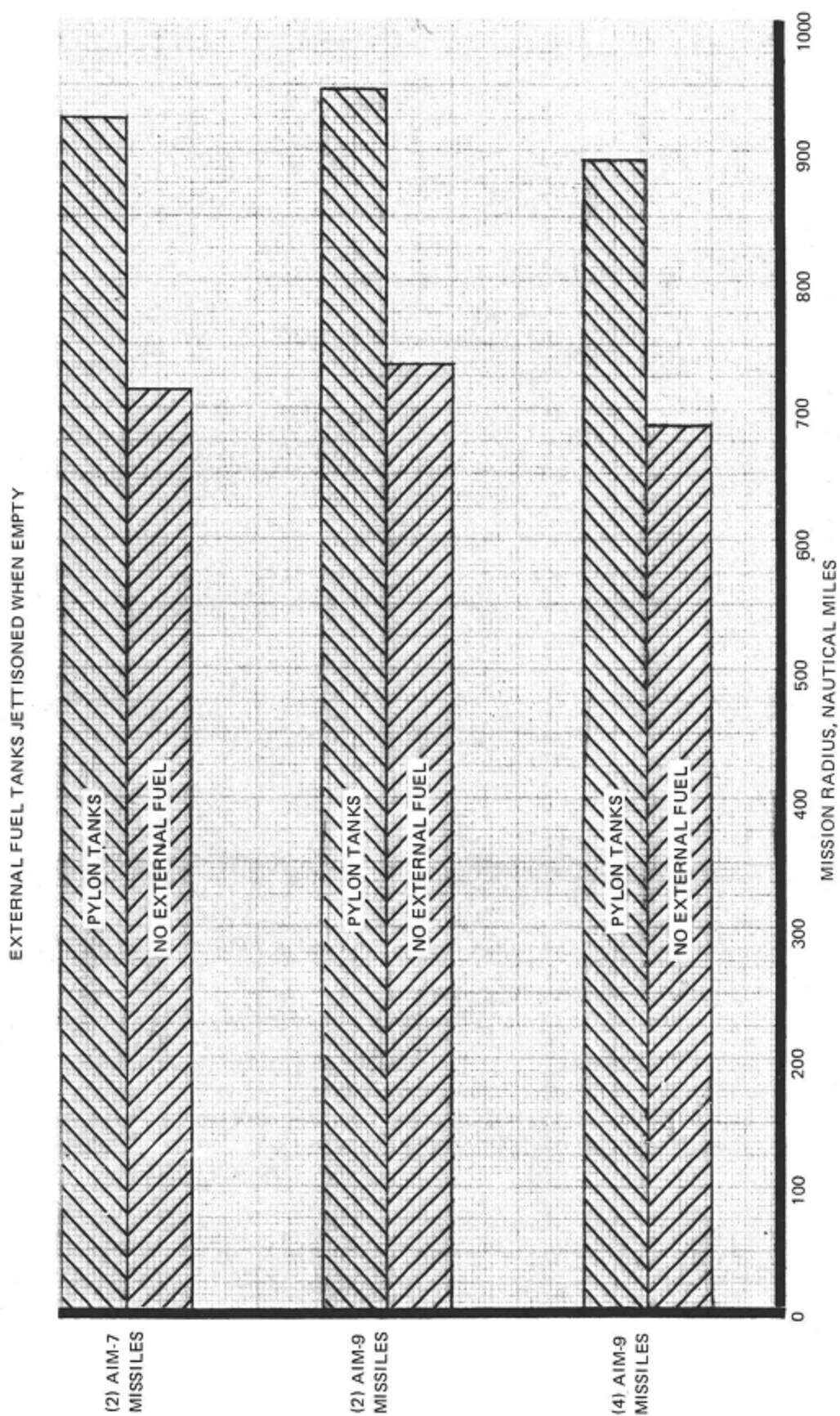
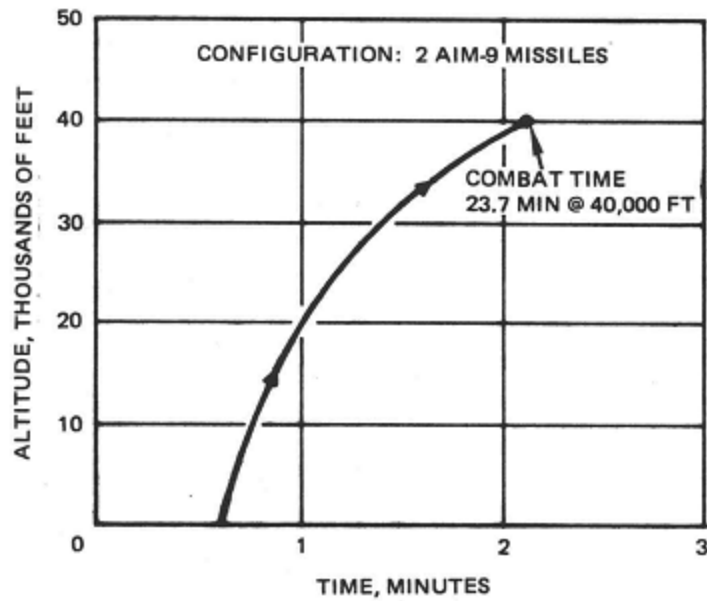


Figure 2-13. Subsonic Area Intercept Mission Performance



MISSION PROFILE:

1. MAXIMUM POWER TAKEOFF AND ACCELERATE TO CLIMB SPEED.
2. MAXIMUM POWER CLIMB TO COMBAT ALTITUDE, $M = 0.90$.
3. COMBAT AT MAXIMUM POWER, $M = 0.90$, FIRE MISSILES AND AMMO.
4. DESCEND TO OPTIMUM CRUISE ALTITUDE. NO TIME, DISTANCE OR FUEL CREDIT.
5. RETURN CRUISE, OPTIMUM MACH AND ALTITUDE.
6. DESCEND TO SEA LEVEL. NO TIME, FUEL OR DISTANCE CREDIT.
7. RESERVES: FUEL FOR 20 MINUTES SEA LEVEL LOITER.

Figure 2-14. Subsonic Point Intercept Mission Profile

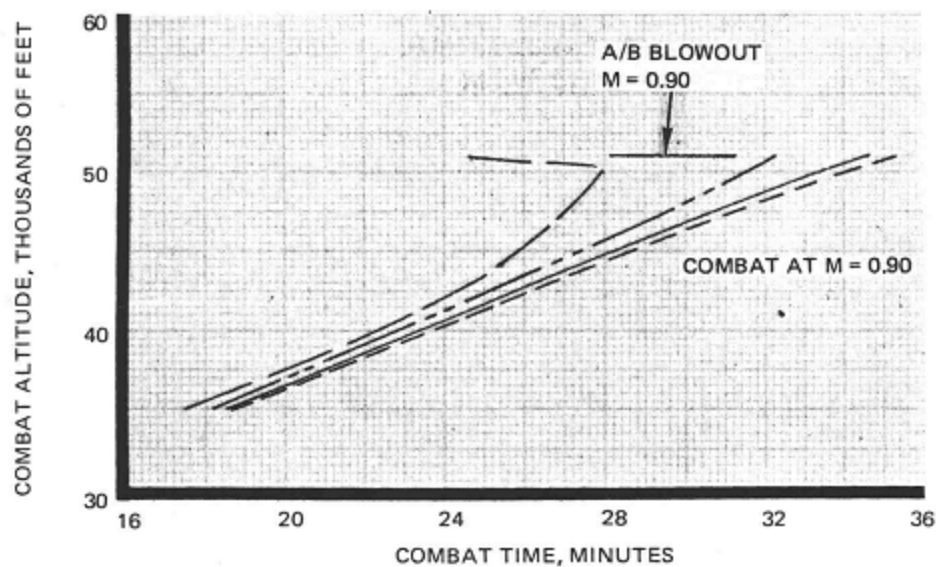
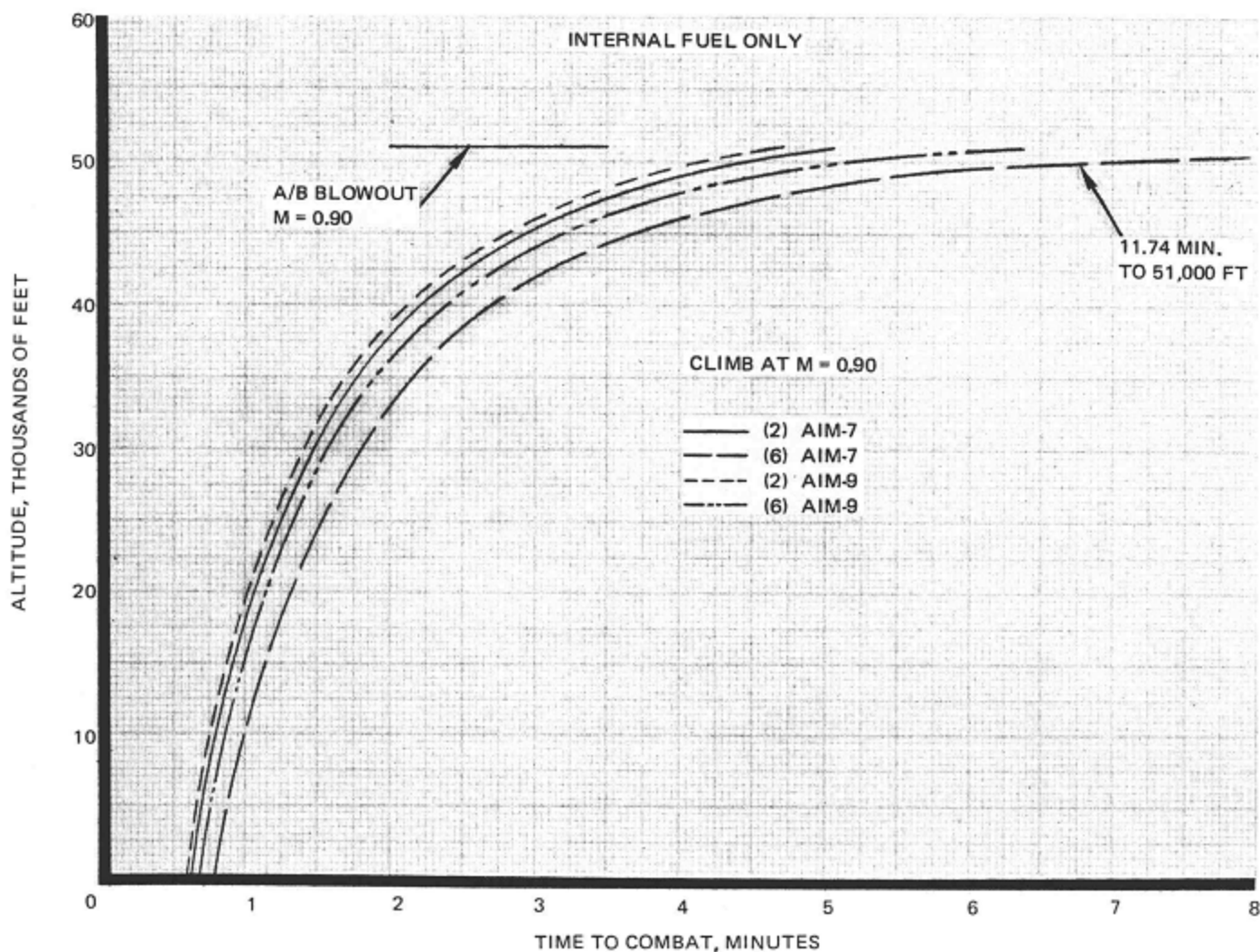
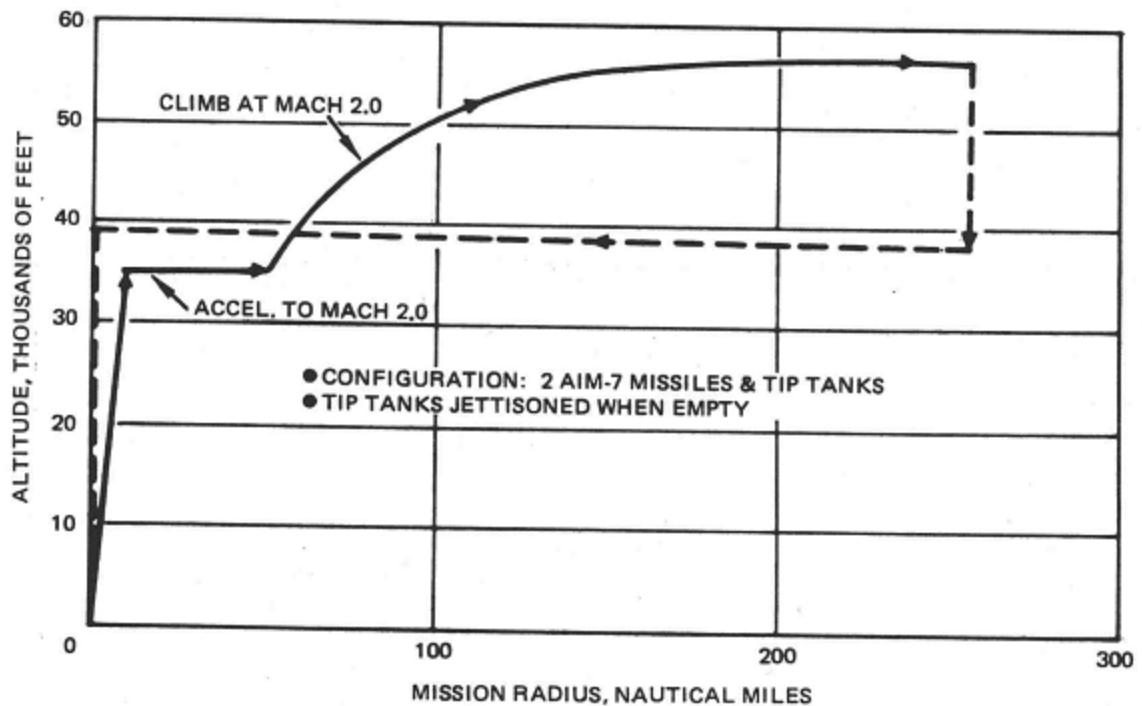


Figure 2-15. Subsonic Point Intercept Mission Performance



MISSION PROFILE:

1. TAKEOFF ALLOWANCE: MAXIMUM POWER TAKEOFF AND ACCELERATION TO CLIMB SPEED.
2. MAXIMUM POWER CLIMB TO 35,000 FEET, $M = 0.90$.
3. MAXIMUM POWER ACCELERATION TO SPECIFIED MACH NUMBER.
4. MAXIMUM POWER CLIMB AT CONSTANT MACH NUMBER TO COMBAT ALTITUDE.
5. COMBAT: 5 MINUTES AT MAXIMUM POWER, FIRE MISSILES AND AMMO.
6. DESCEND TO OPTIMUM CRUISE ALTITUDE, NO FUEL, TIME OR DISTANCE CREDIT.
7. RETURN CRUISE, OPTIMUM MACH AND ALTITUDE.
8. DESCEND TO SEA LEVEL, NO FUEL, TIME OR DISTANCE CREDIT.
9. RESERVES: FUEL FOR 20 MINUTES SEA LEVEL LOITER.

Figure 2-16. Supersonic High Altitude Intercept Mission Profile

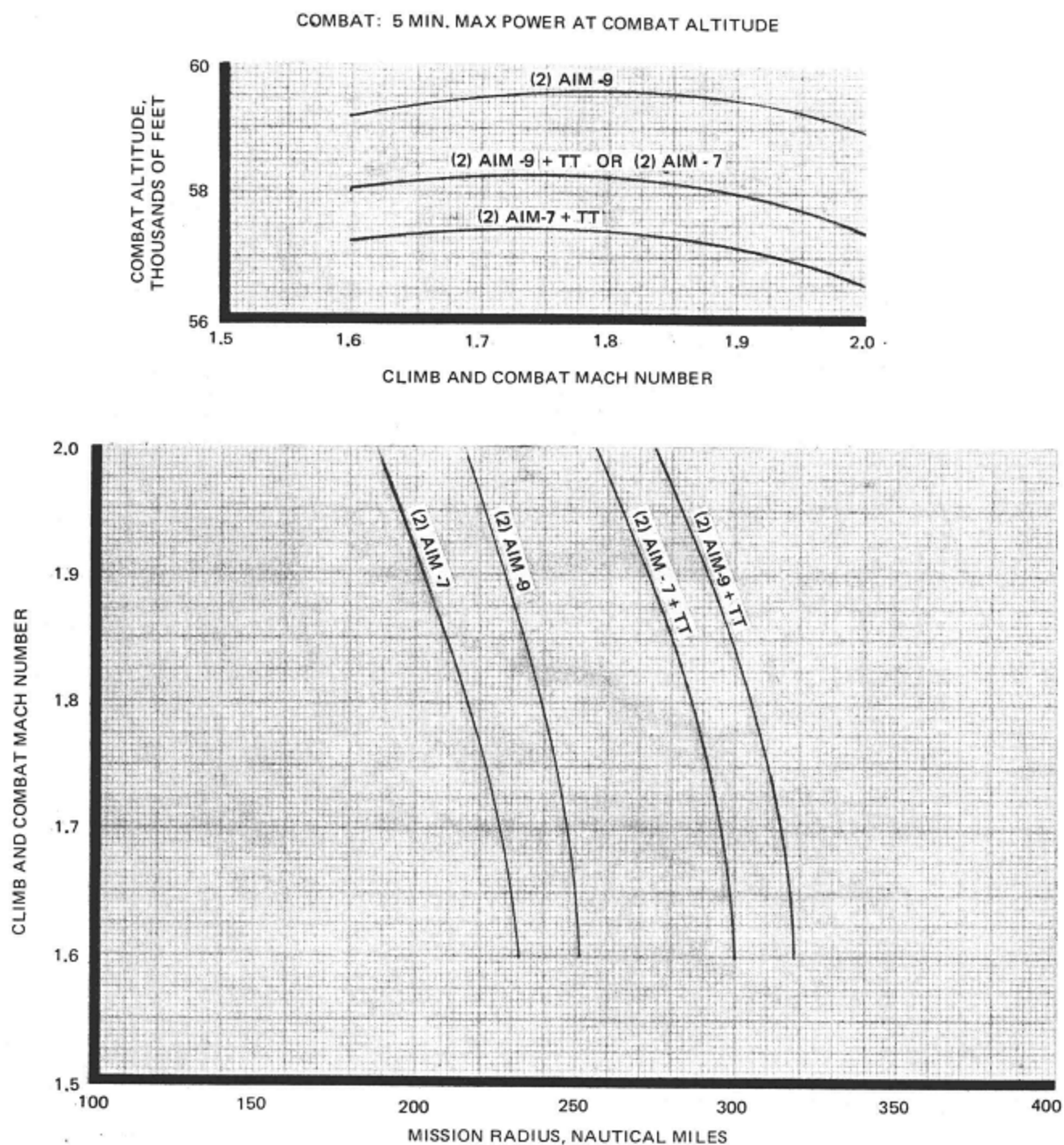
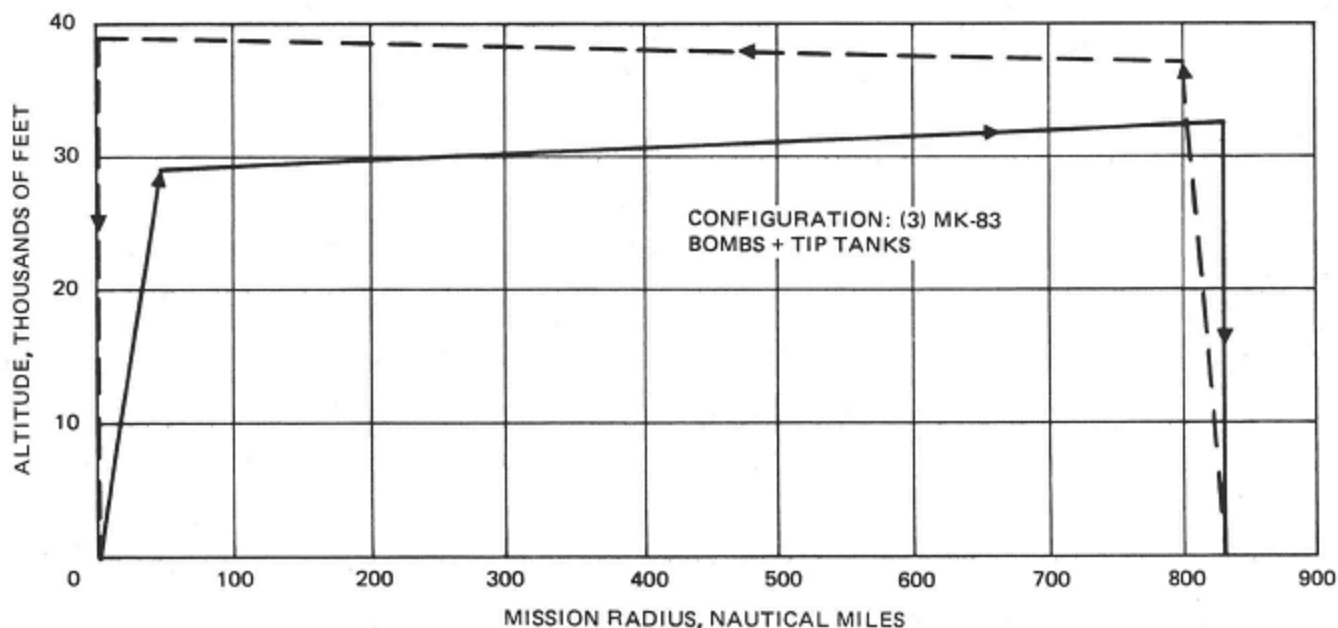


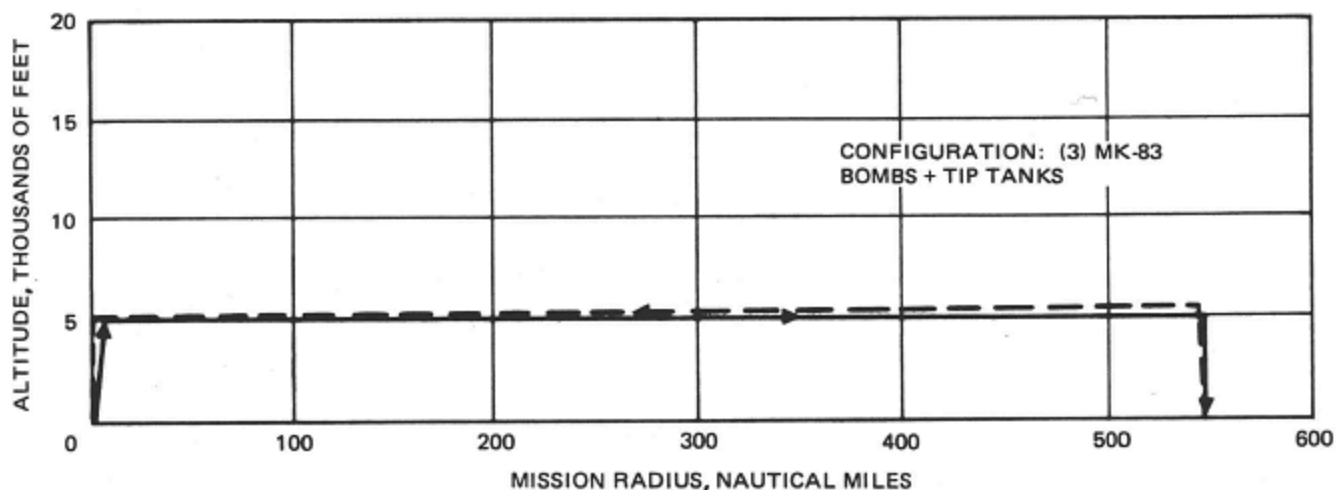
Figure 2-17. Supersonic High Altitude Intercept Mission Performance



MISSION PROFILE:

1. TAKEOFF ALLOWANCE: FUEL FOR 6.0 MINUTES AT $T/W = 0.20$ + FUEL TO TAKEOFF AND ACCELERATE TO CLIMB SPEED AT MAXIMUM POWER.
2. INTERMEDIATE POWER CLIMB TO OPTIMUM CRUISE ALTITUDE.
3. OUTBOUND CRUISE, OPTIMUM MACH AND ALTITUDE. EXTERNAL FUEL TANKS JETTISONED WHEN EMPTY.
4. DESCEND TO SEA LEVEL. NO TIME, DISTANCE, OR FUEL CREDIT.
5. COMBAT: 5 MINUTES AT INTERMEDIATE POWER AT SEA LEVEL, $M = 0.70$. DROP BOMBS AND FIRE AMMO.
6. INTERMEDIATE POWER CLIMB TO OPTIMUM CRUISE ALTITUDE.
7. RETURN CRUISE, OPTIMUM MACH AND ALTITUDE.
8. DESCEND TO SEA LEVEL. NO TIME, DISTANCE OR FUEL CREDIT.
9. RESERVES: FUEL FOR 20 MINUTES SEA LEVEL LOITER.

Figure 2-18. Counterattack Mission: Hi-Lo-Hi Profile



MISSION PROFILE:

1. TAKEOFF ALLOWANCE: FUEL FOR 6.0 MINUTES AT $T/W = 0.20$ + FUEL TO TAKEOFF AND ACCELERATE TO CLIMB SPEED AT MAXIMUM POWER.
2. INTERMEDIATE POWER CLIMB TO 5,000 FEET ALTITUDE.
3. OUTBOUND CRUISE AT 5,000 FEET, OPTIMUM MACH, EXTERNAL FUEL TANKS JETTISONED WHEN EMPTY.
4. DESCEND TO SEA LEVEL, NO TIME, DISTANCE OR FUEL CREDIT.
5. COMBAT: 5 MINUTES AT INTERMEDIATE POWER AT SEA LEVEL, $M = 0.70$, DROP BOMBS AND FIRE AMMO.
6. INTERMEDIATE POWER CLIMB TO 5,000 FEET ALTITUDE.
7. RETURN CRUISE AT 5,000 FEET, OPTIMUM MACH.
8. DESCEND TO SEA LEVEL, NO TIME, DISTANCE OR FUEL CREDIT.
9. RESERVES: FUEL FOR 20 MINUTES SEA LEVEL LOITER.

Figure 2-19. Counterattack Mission: Lo-Lo-Lo Profile

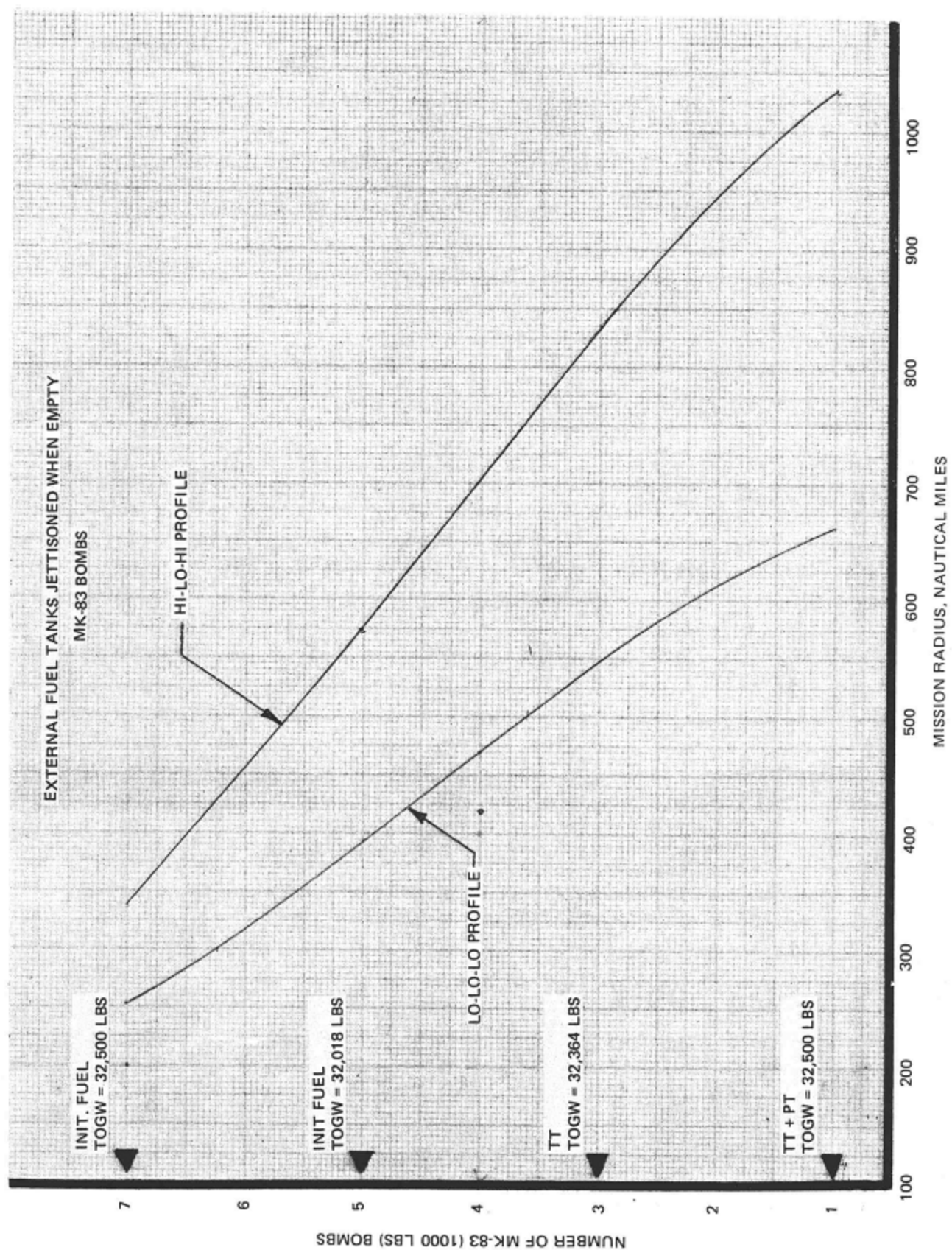
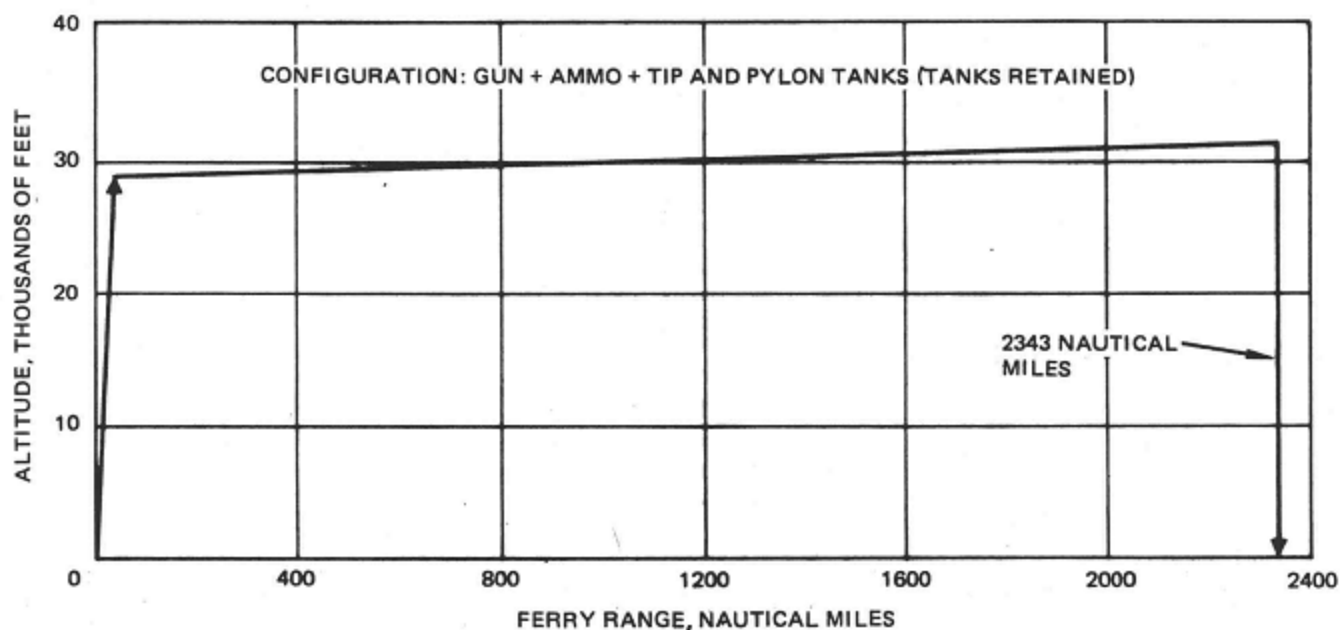


Figure 2-20. Counterattack Mission Performance



MISSION PROFILE:

1. TAKEOFF ALLOWANCE: FUEL FOR 6 MINUTES AT $T/W = 0.20$ + FUEL TO TAKEOFF AND ACCELERATE TO CLIMB SPEED AT MAXIMUM POWER.
2. INTERMEDIATE POWER CLIMB TO OPTIMUM CRUISE ALTITUDE.
3. CRUISE, OPTIMUM MACH AND ALTITUDE.
4. DESCEND TO SEA LEVEL, NO TIME, DISTANCE OR FUEL CREDIT.
5. RESERVES: FUEL FOR 20 MINUTES SEA LEVEL LOITER.

Figure 2-21. Ferry Mission

2.6 SPECIFIC EXCESS POWER

The specific excess power (SEP) available for acceleration, climb, or turning maneuvers throughout the flight envelope is an important measure of an aircraft's combat capability. SEP is defined as:

$$\text{SEP} = \frac{\text{Velocity (Thrust-Drag)}}{\text{Weight}}, \text{ ft/sec}$$

The SEP capabilities of the Lancer at g values of one through seven are presented in Figures 2-22 through 2-42, with 50% of internal fuel, for a clean configuration and for configurations with tip-mounted AIM-7 Sparrow or AIM-9 Sidewinder missiles. Maximum afterburning power and automatic maneuvering flaps are assumed.

2.7 STEADY STATE AND MAXIMUM INSTANTANEOUS LOAD FACTOR CAPABILITY

Steady state (thrust equals drag) and maximum instantaneous (maximum usable lift) load factor contours for the Lancer are presented in Figures 2-43 through 2-48, for the same conditions, weights, and configurations as for the SEP plots of Figures 2-22 through 2-42. The steady state contours correspond to zero SEP values shown in the SEP plots. The maximum instantaneous contours show the altitude that can be attained with maximum usable lift at each Mach number and load factor.

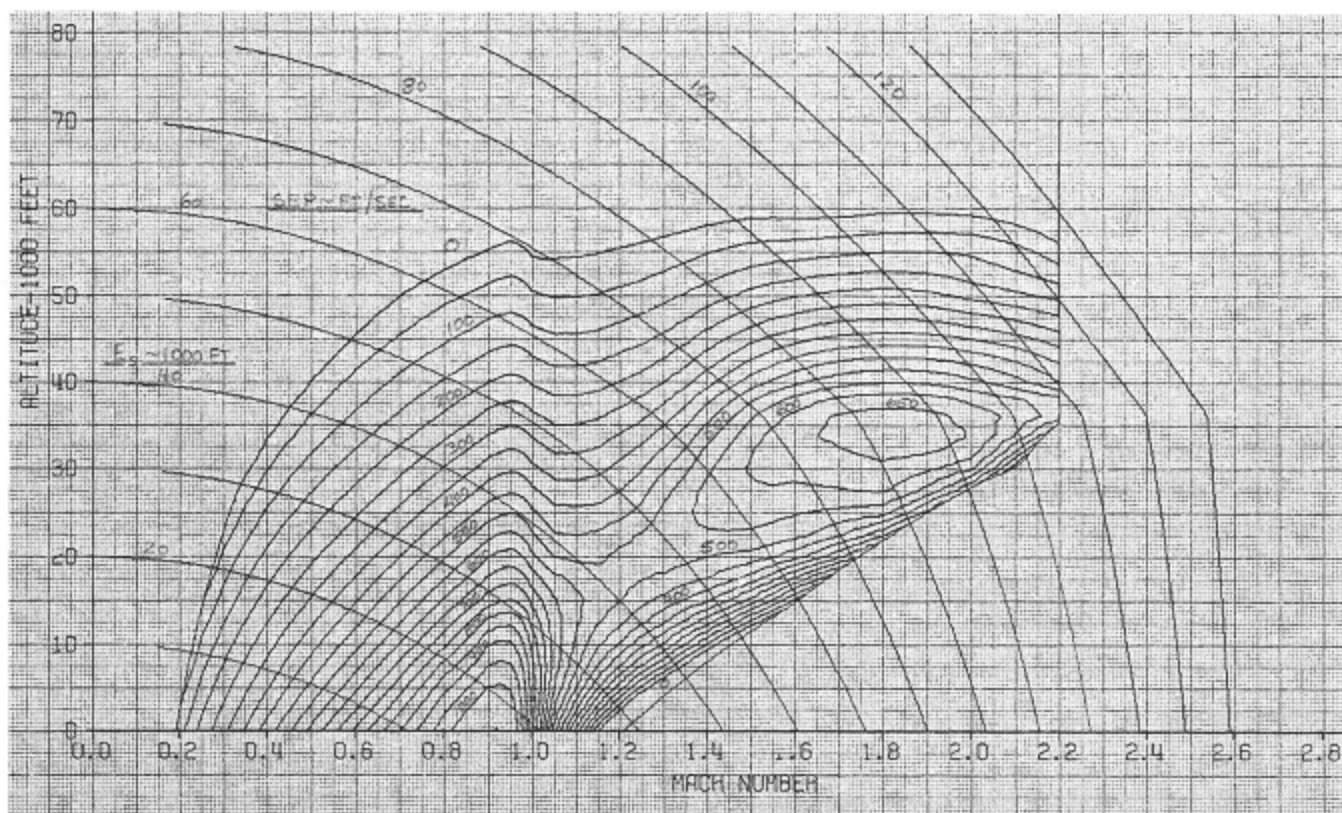


Figure 2-22. Specific Excess Power at 1G, Clean Airplane

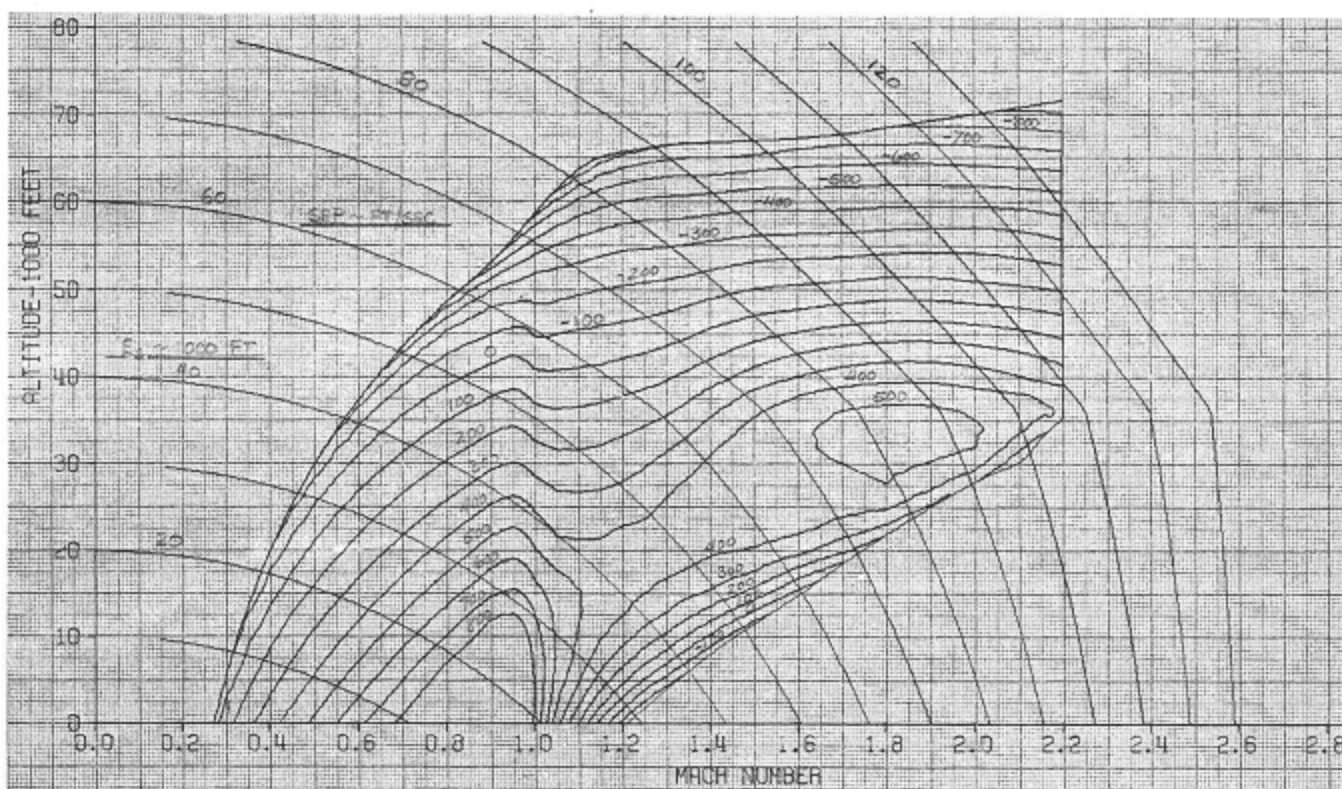


Figure 2-23. Specific Excess Power at 2G, Clean Airplane

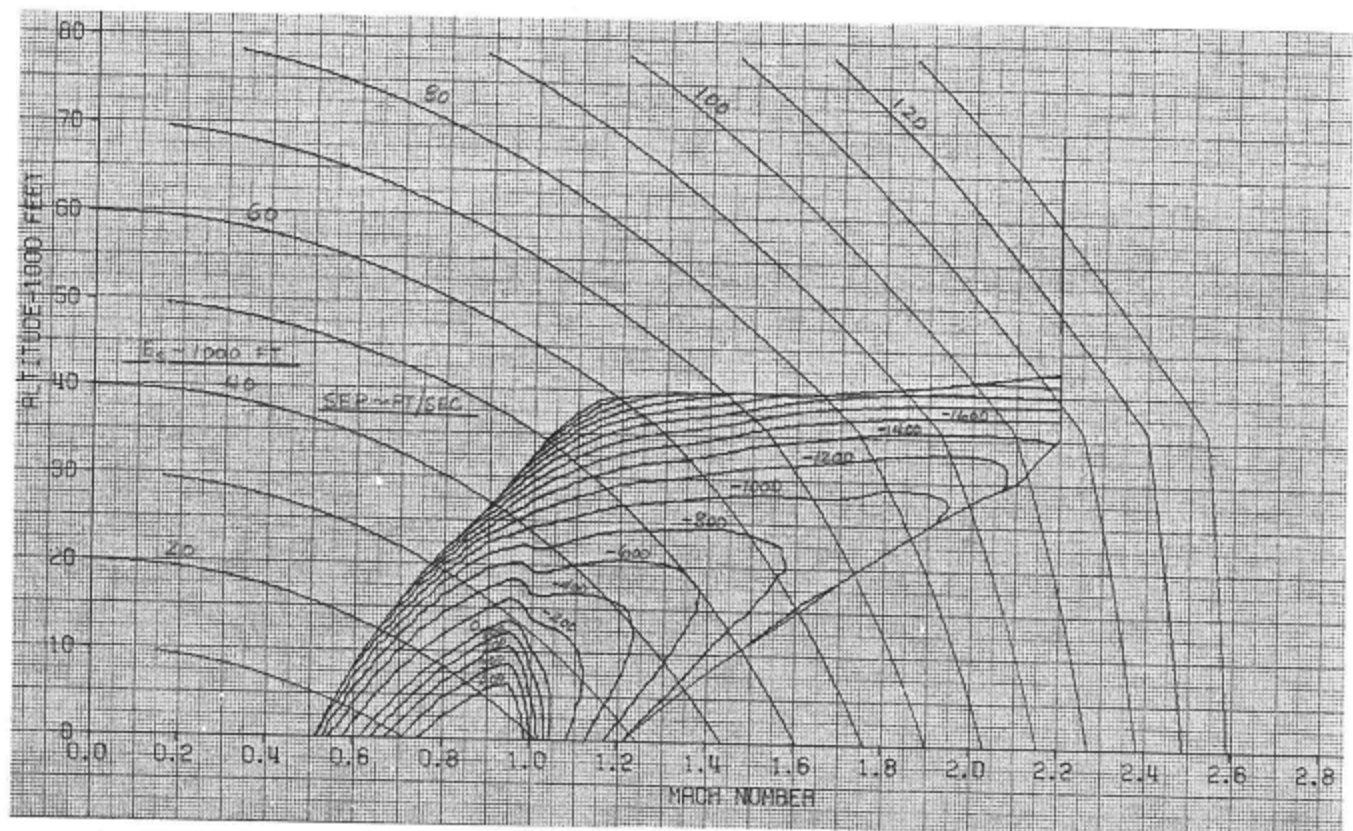


Figure 2-28. Specific Excess Power at 7G, Clean Airplane

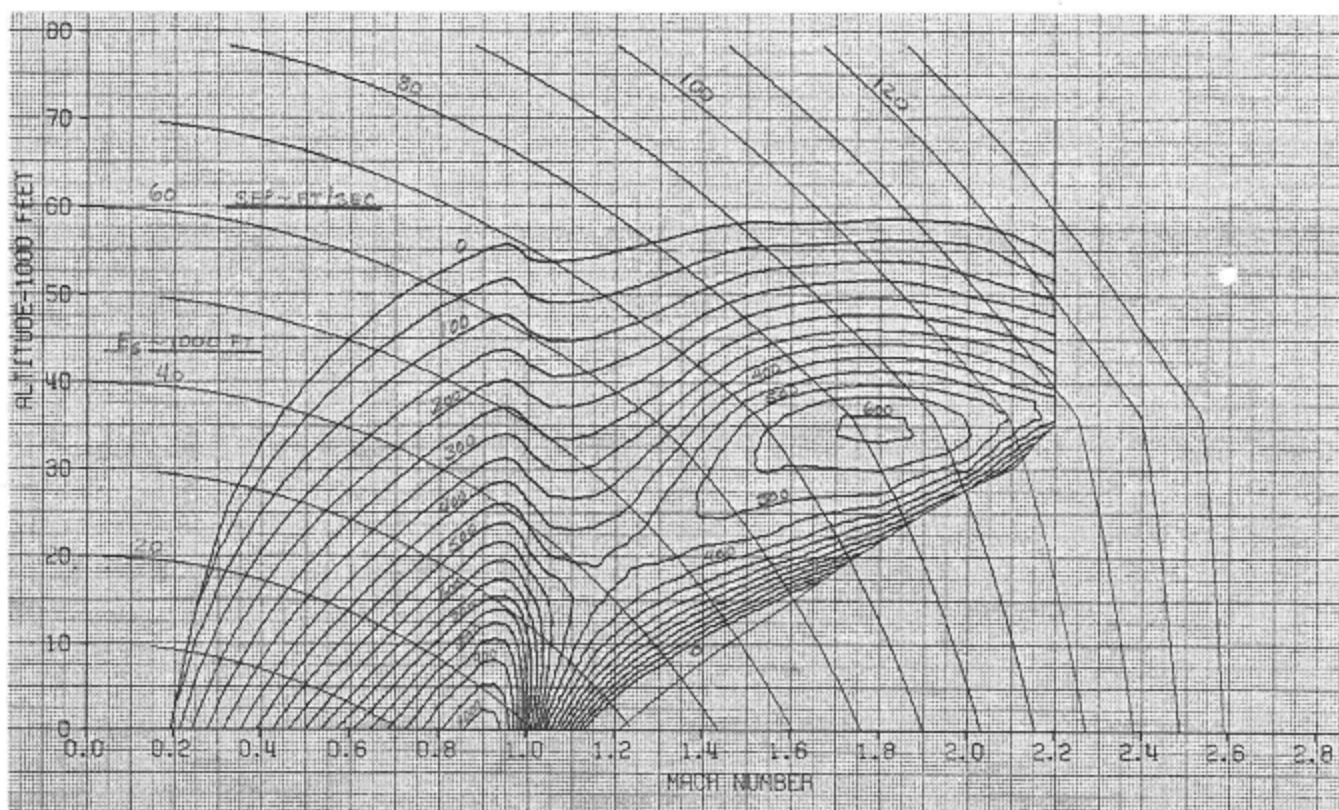


Figure 2-29. Specific Excess Power at 1G, (2) Tip AIM-9

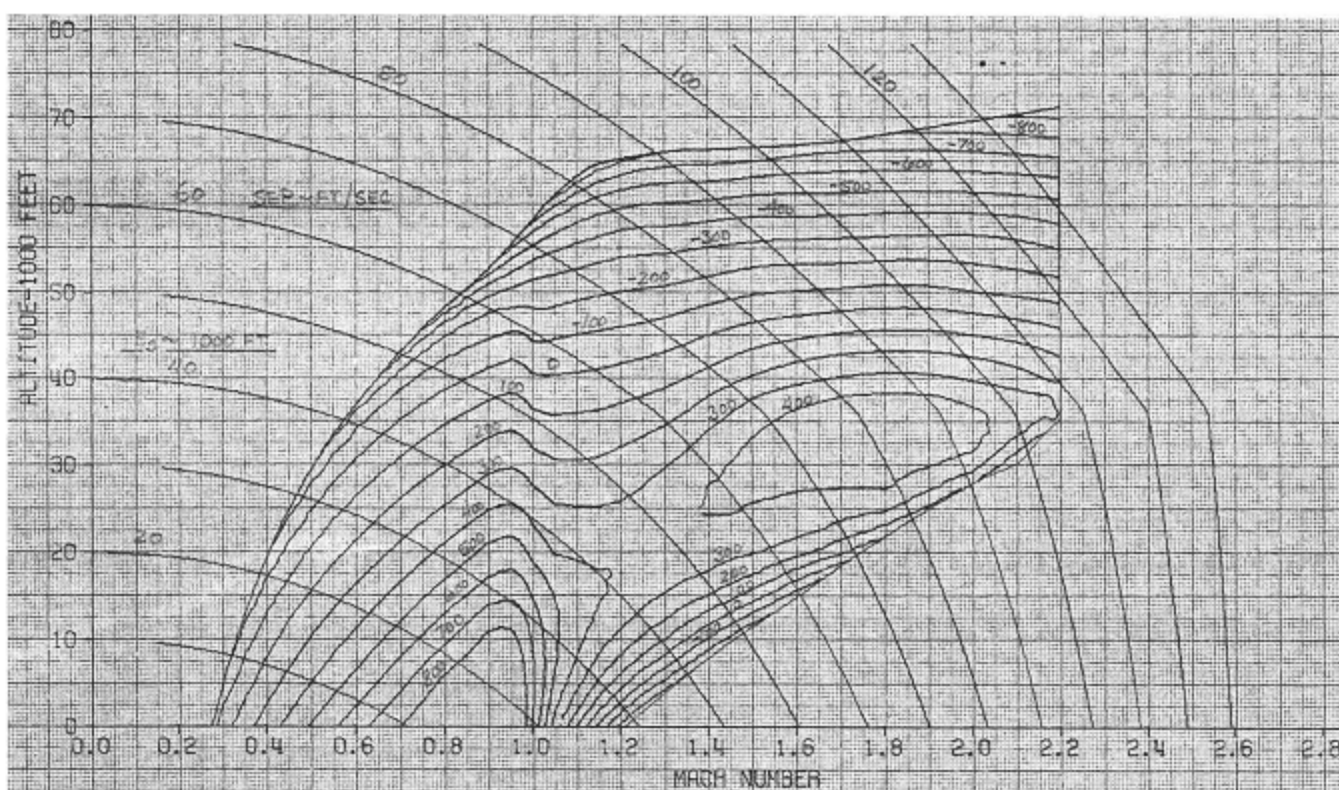


Figure 2-30. Specific Excess Power at 2G, (2) Tip AIM-9

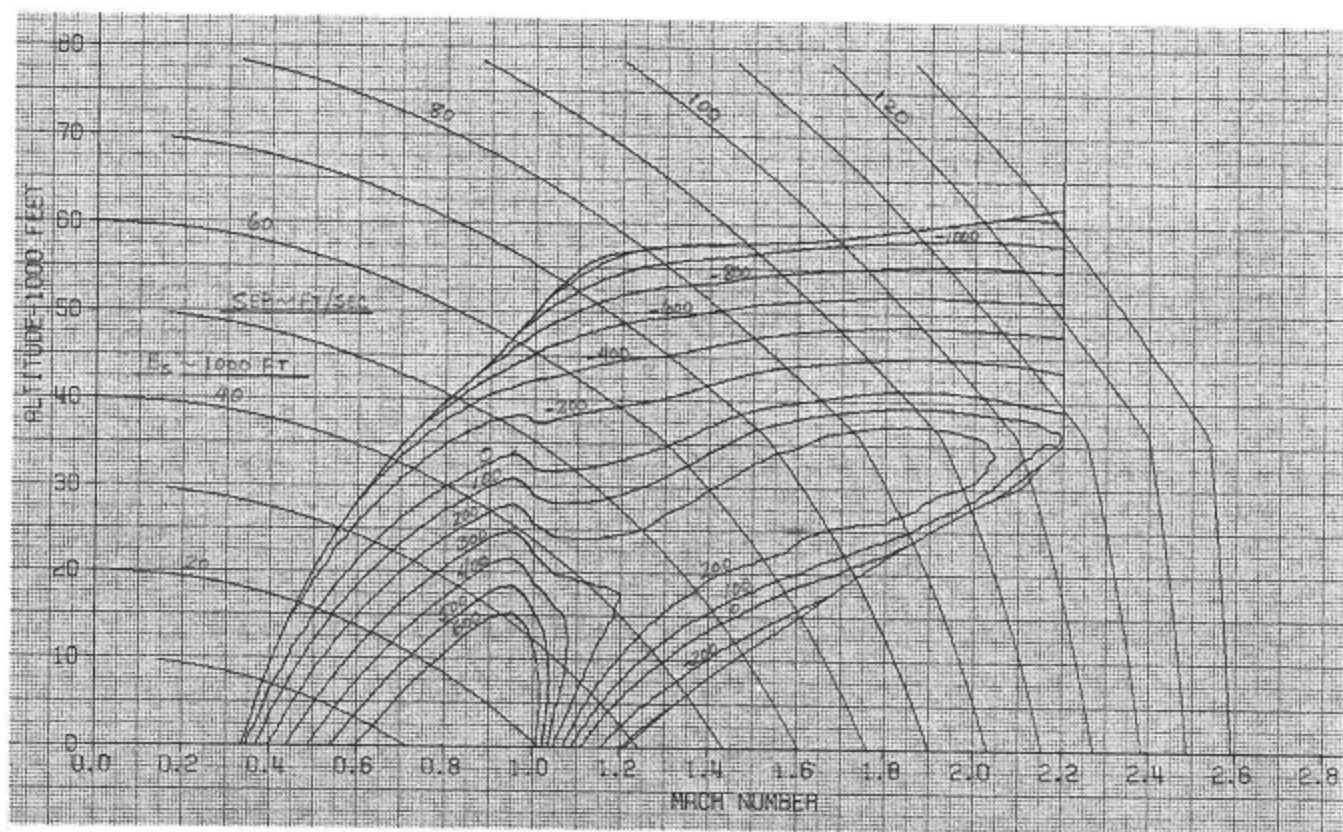


Figure 2-31. Specific Excess Power at 3G, (2) Tip AIM-9

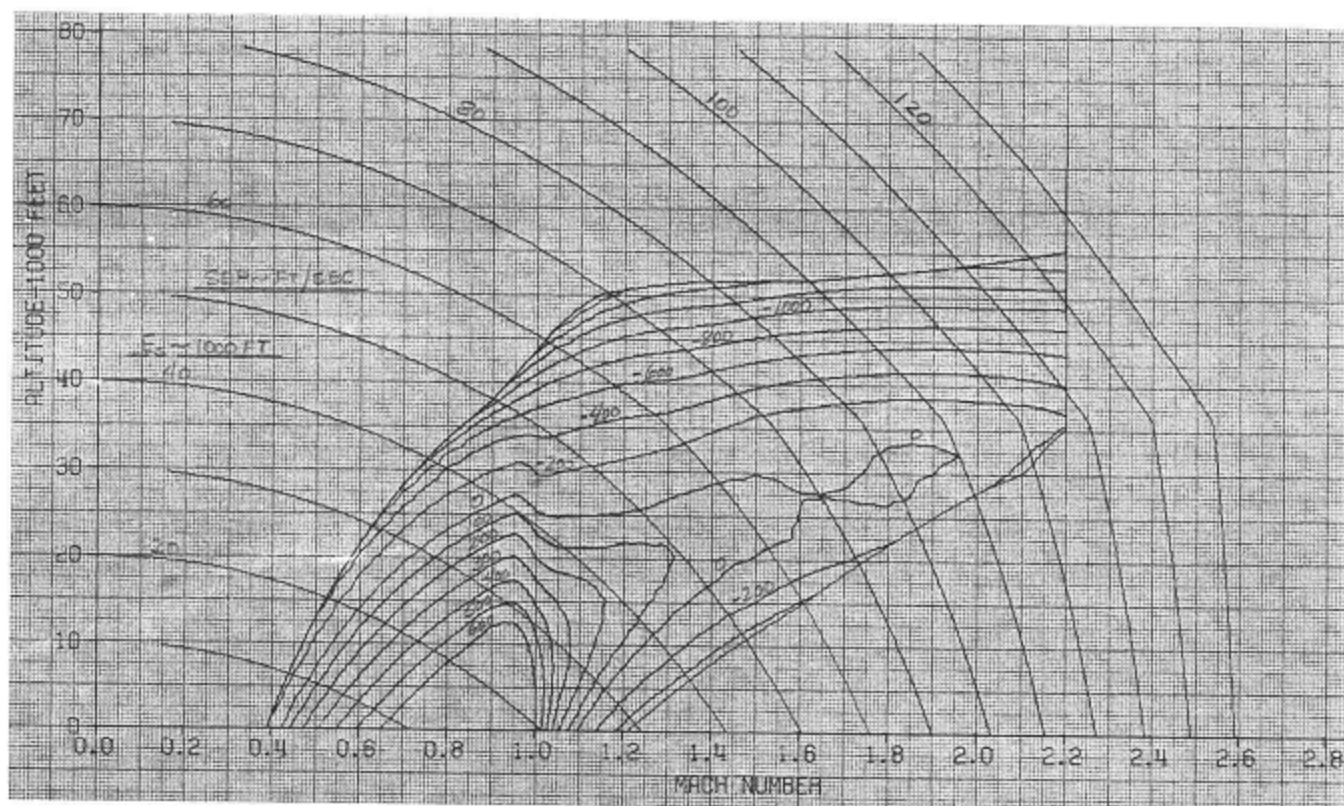


Figure 2-32. Specific Excess Power at 4G, (2) Tip AIM-9

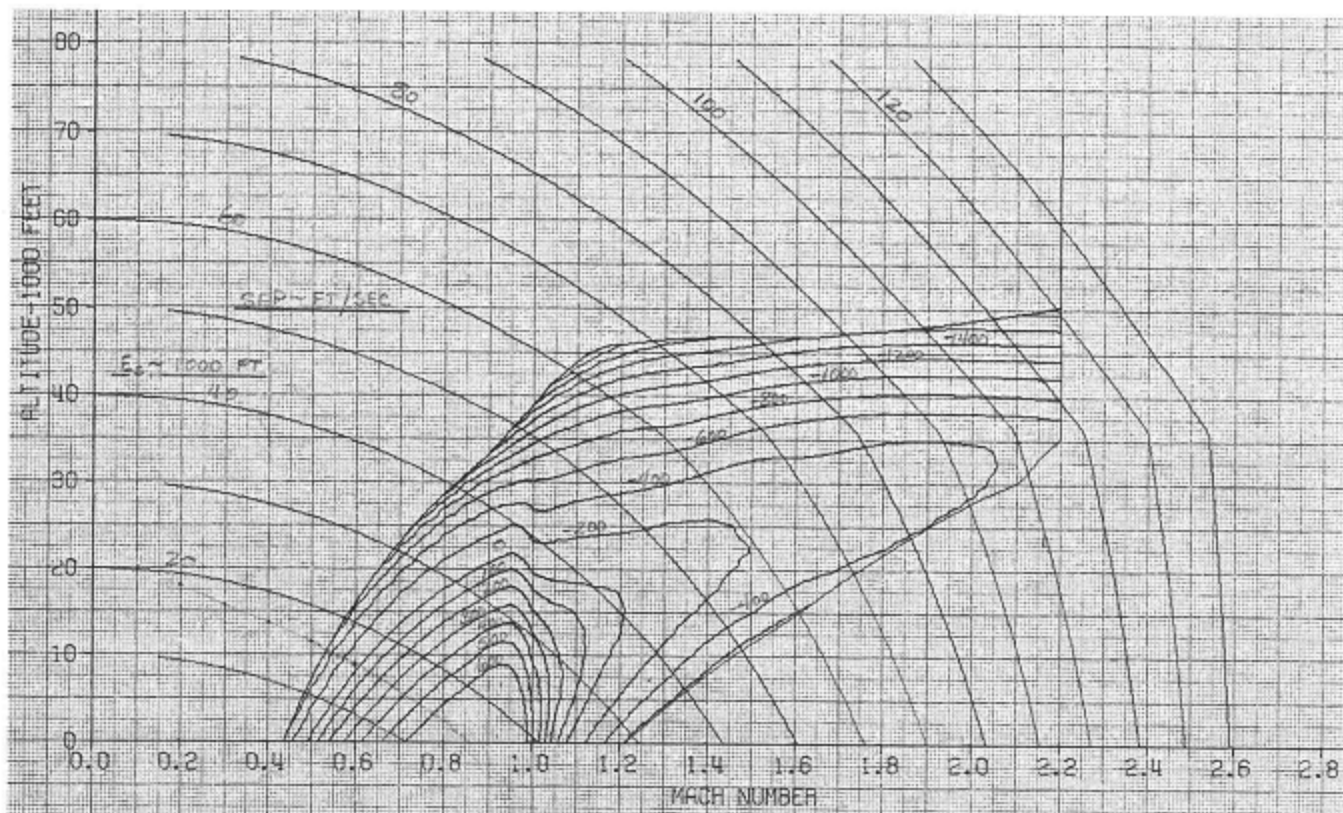


Figure 2-33. Specific Excess Power at 5G, (2) Tip AIM-9

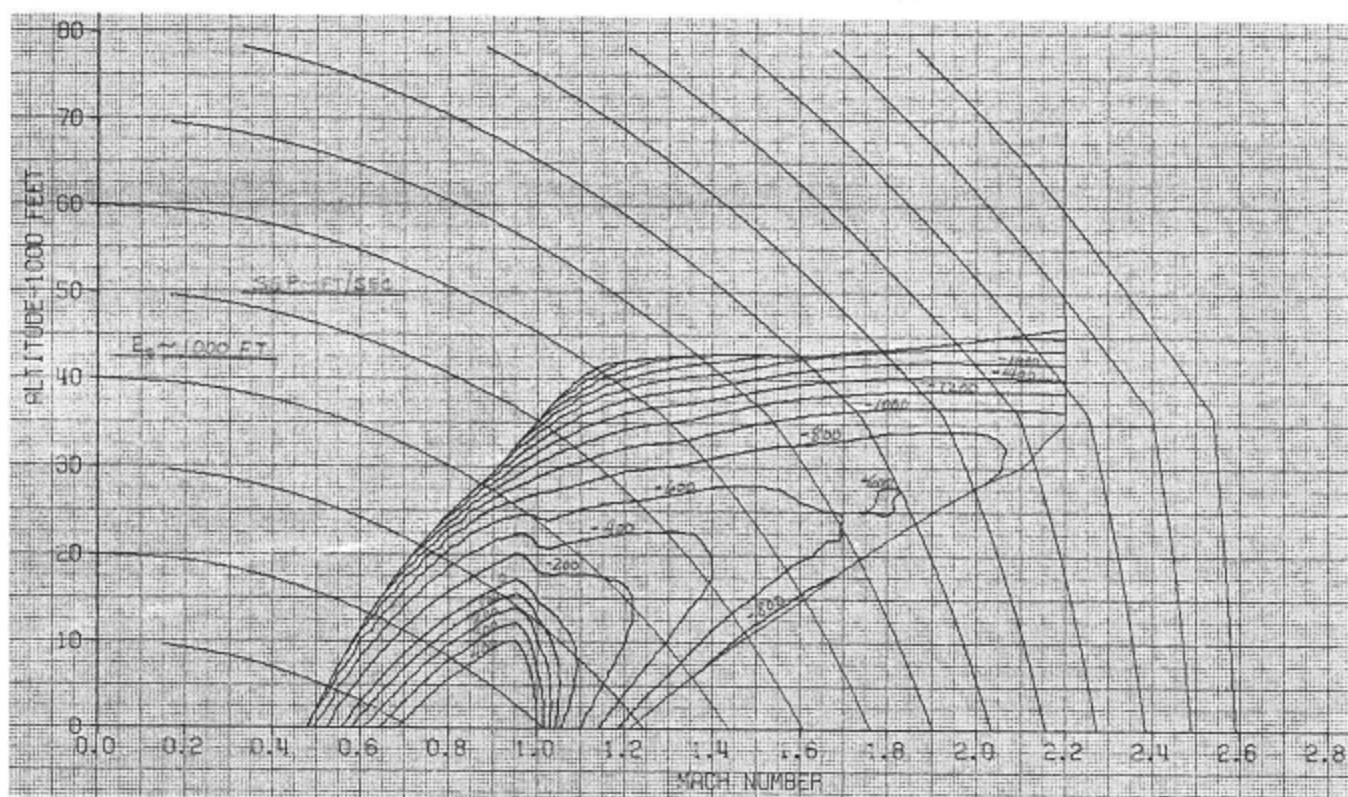


Figure 2-34. Specific Excess Power at 6G, (2) Tip AIM-9

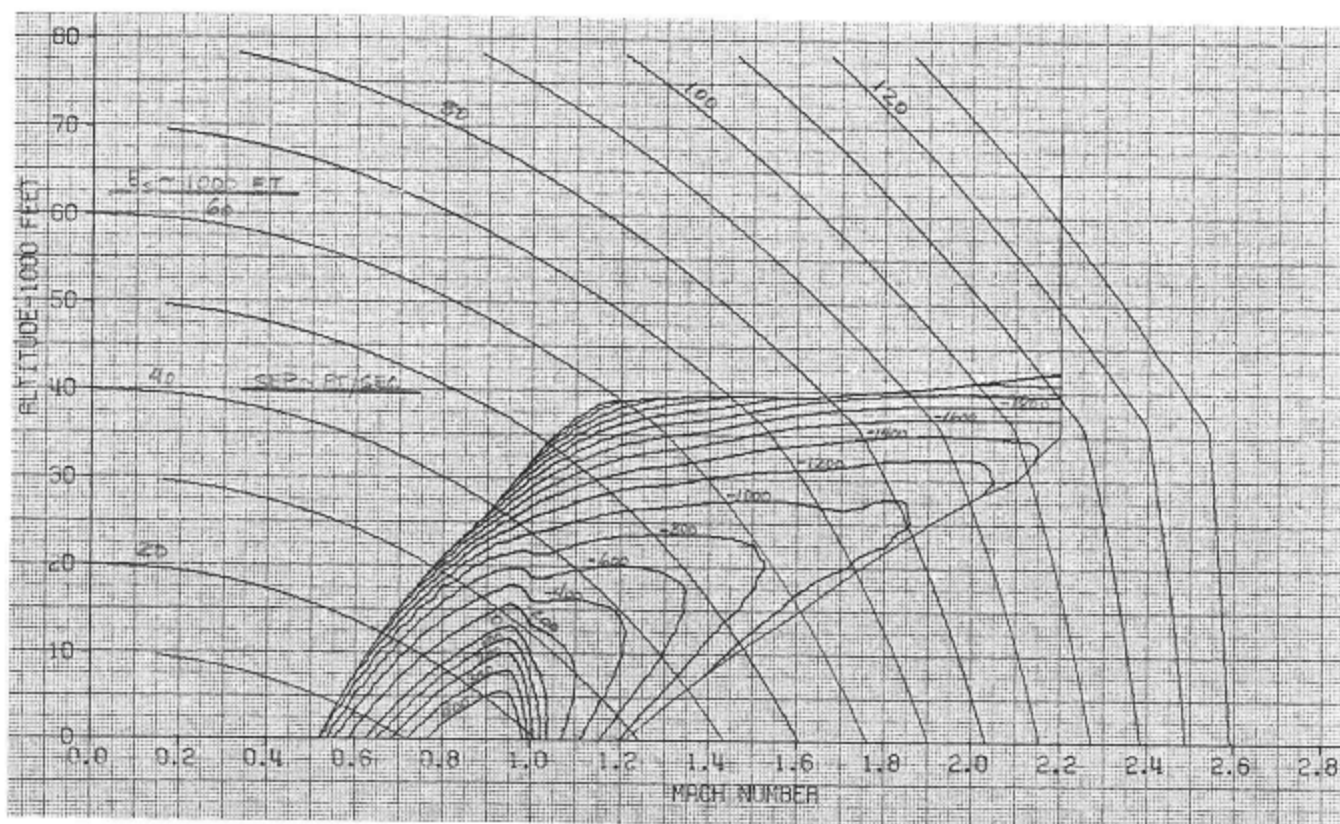


Figure 2-35. Specific Excess Power at 7G, (2) Tip AIM-9

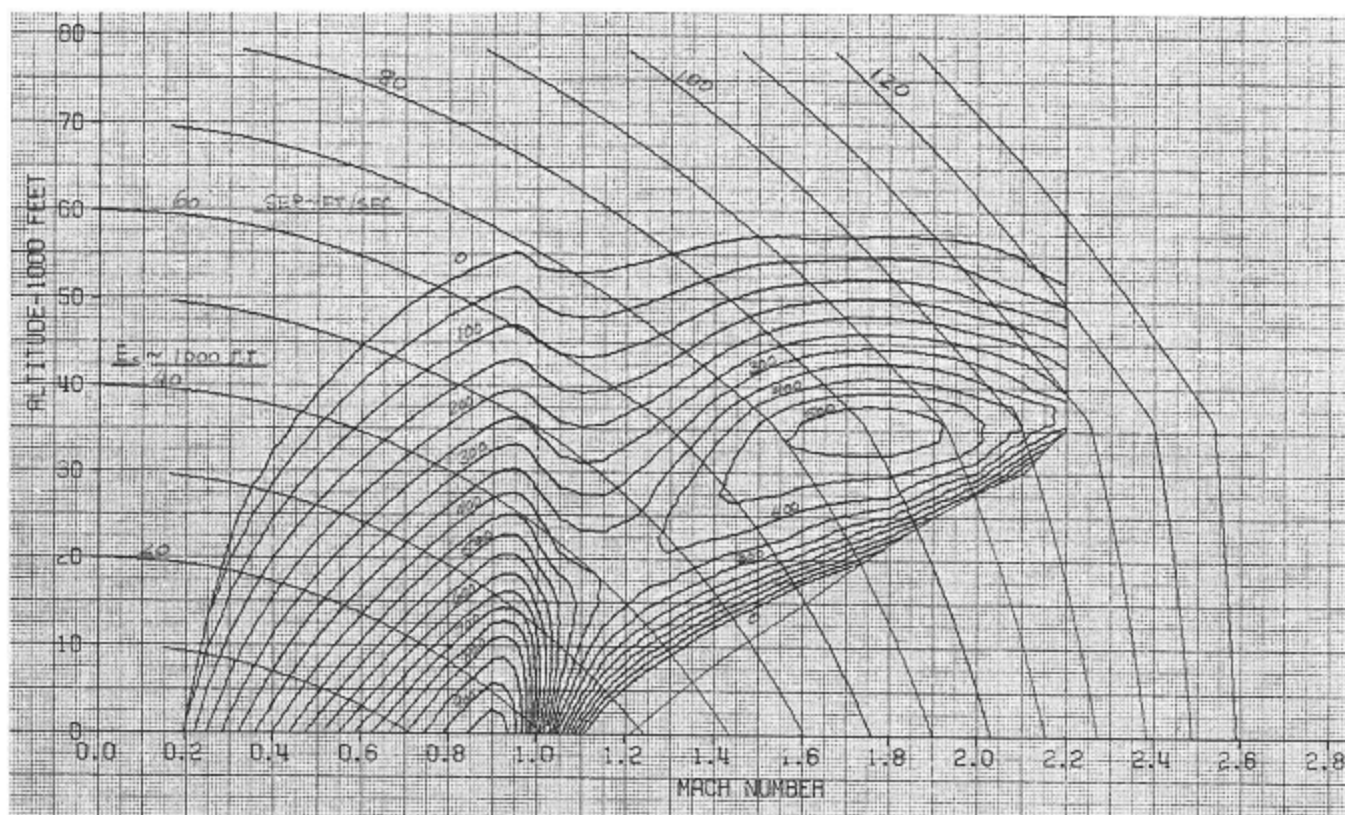


Figure 2-36. Specific Excess Power at 1G, (2) Tip AIM-7

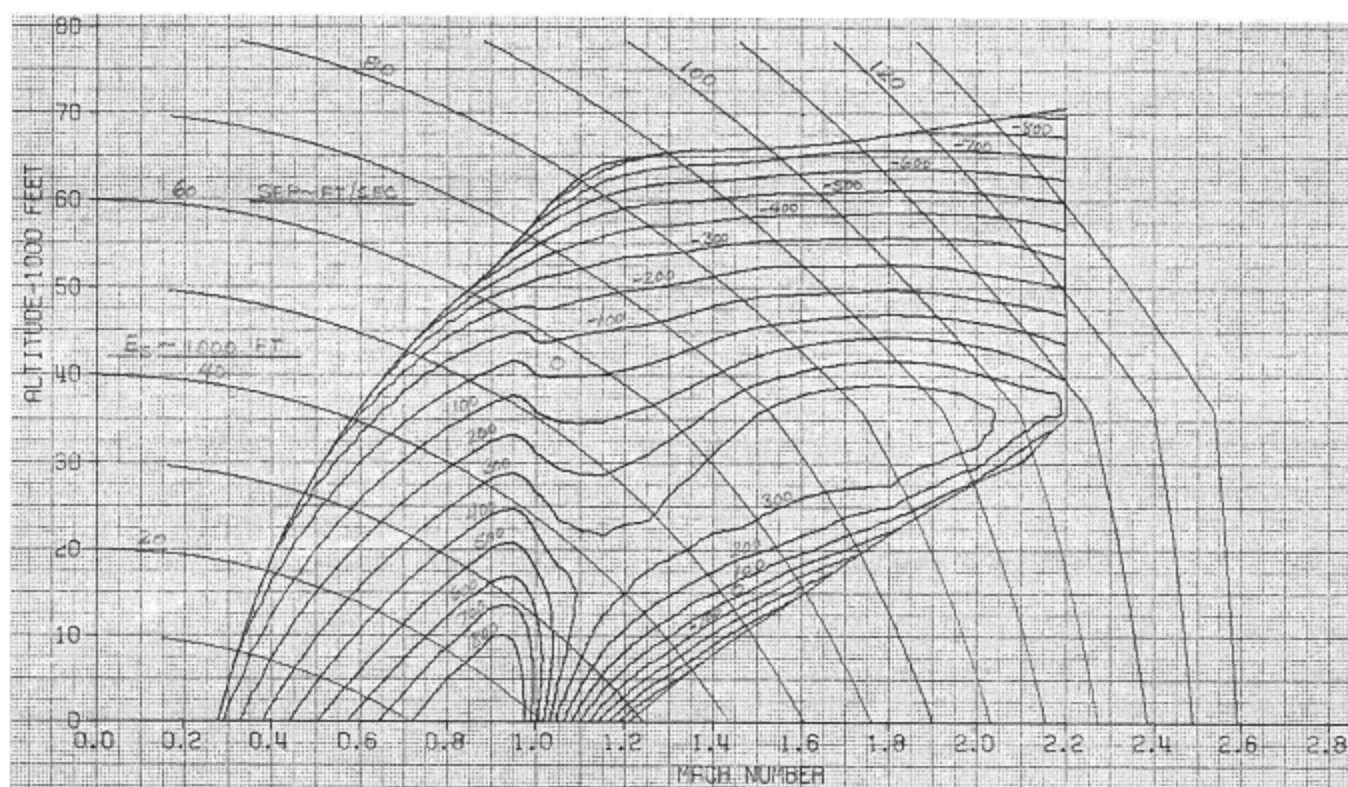


Figure 2-37. Specific Excess Power at 2G, (2) Tip AIM-7

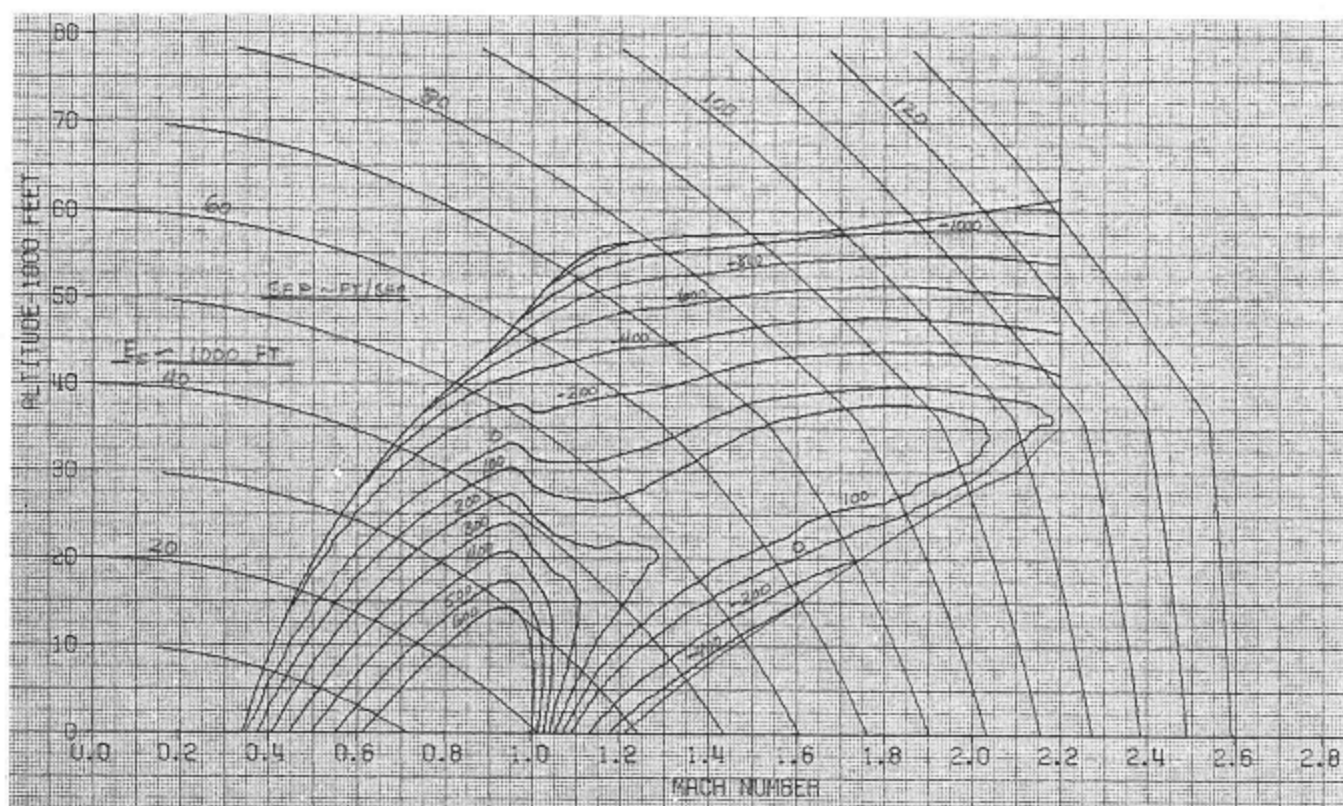


Figure 2-38. Specific Excess Power at 3G, (2) Tip AIM-7

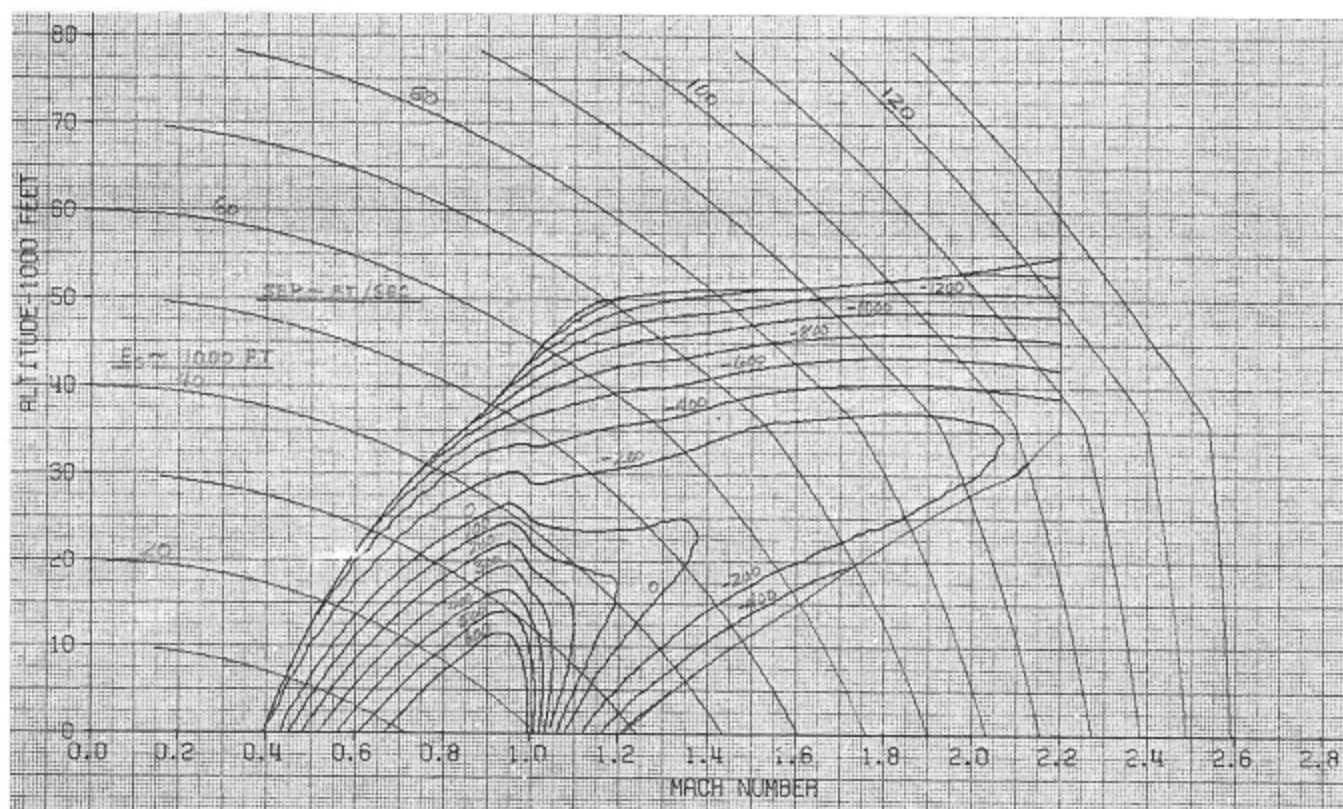


Figure 2-39. Specific Excess Power at 4G, (2) Tip AIM-7

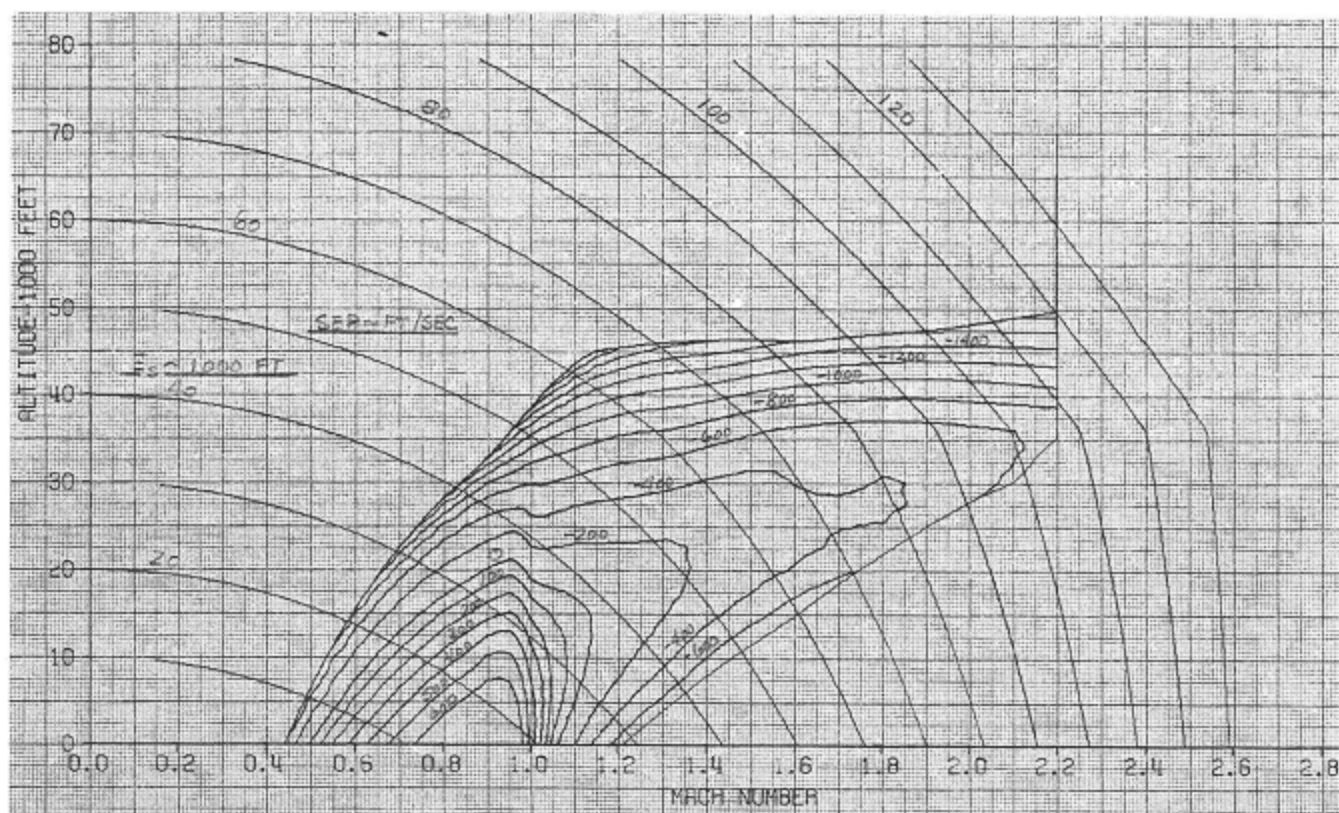


Figure 2-40. Specific Excess Power at 5G, (2) Tip AIM-7

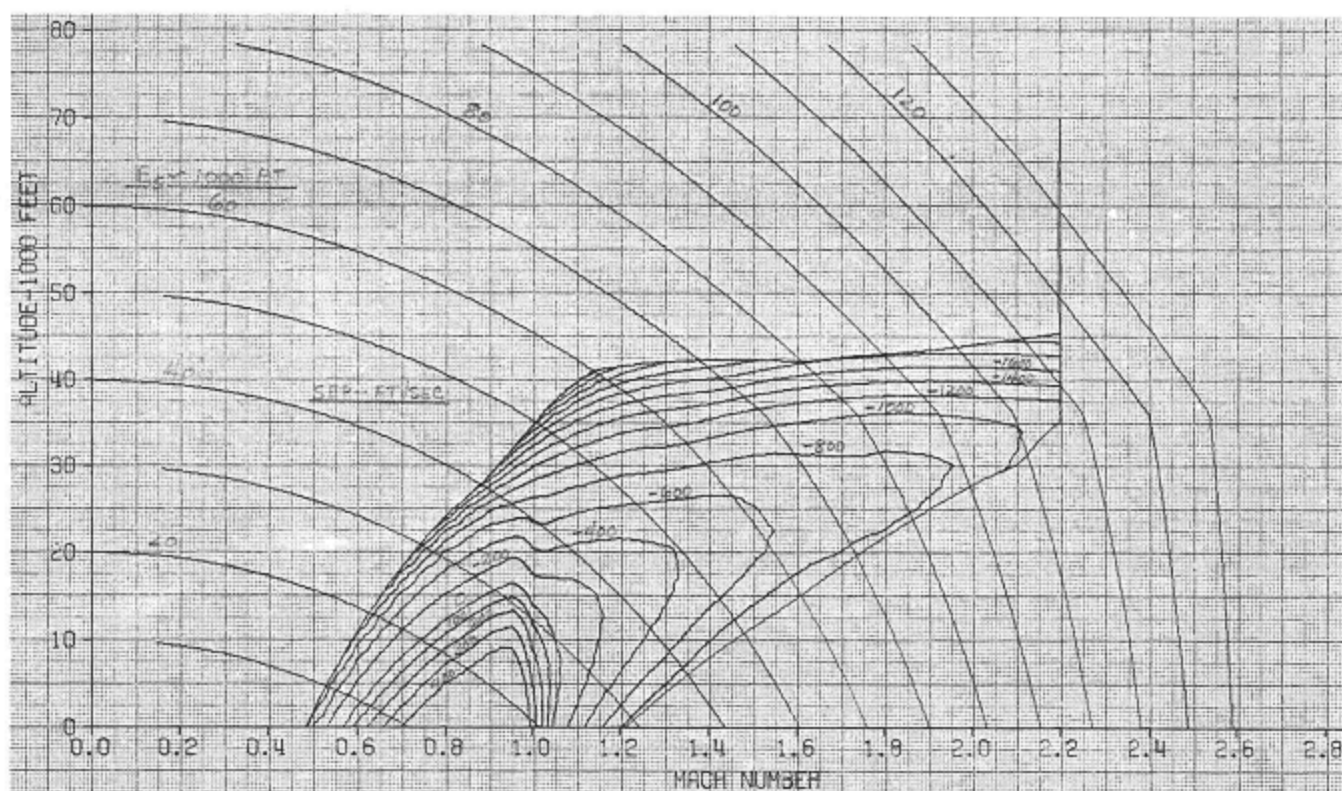


Figure 2-41. Specific Excess Power at 6G, (2) Tip AIM-7

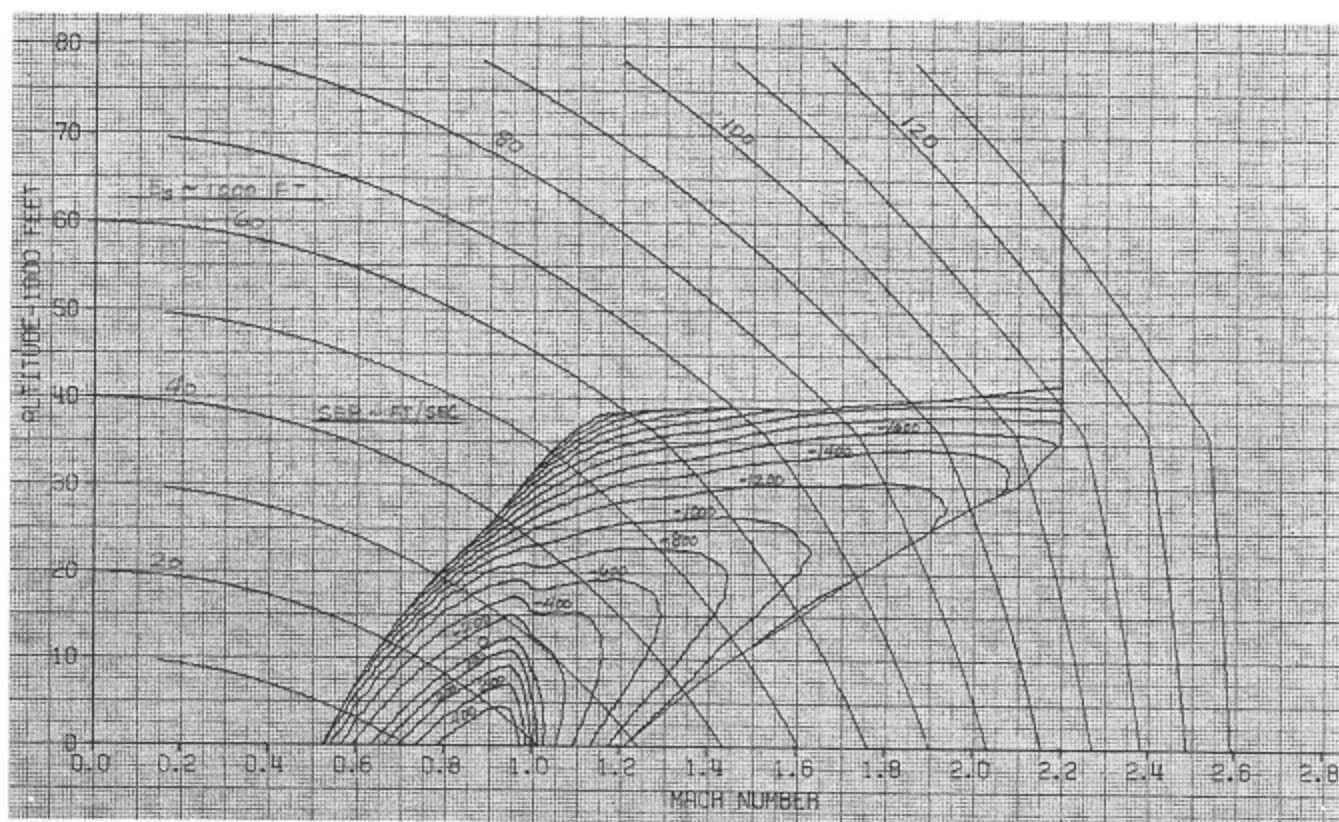


Figure 2-42. Specific Excess Power at 7G, (2) Tip AIM-7

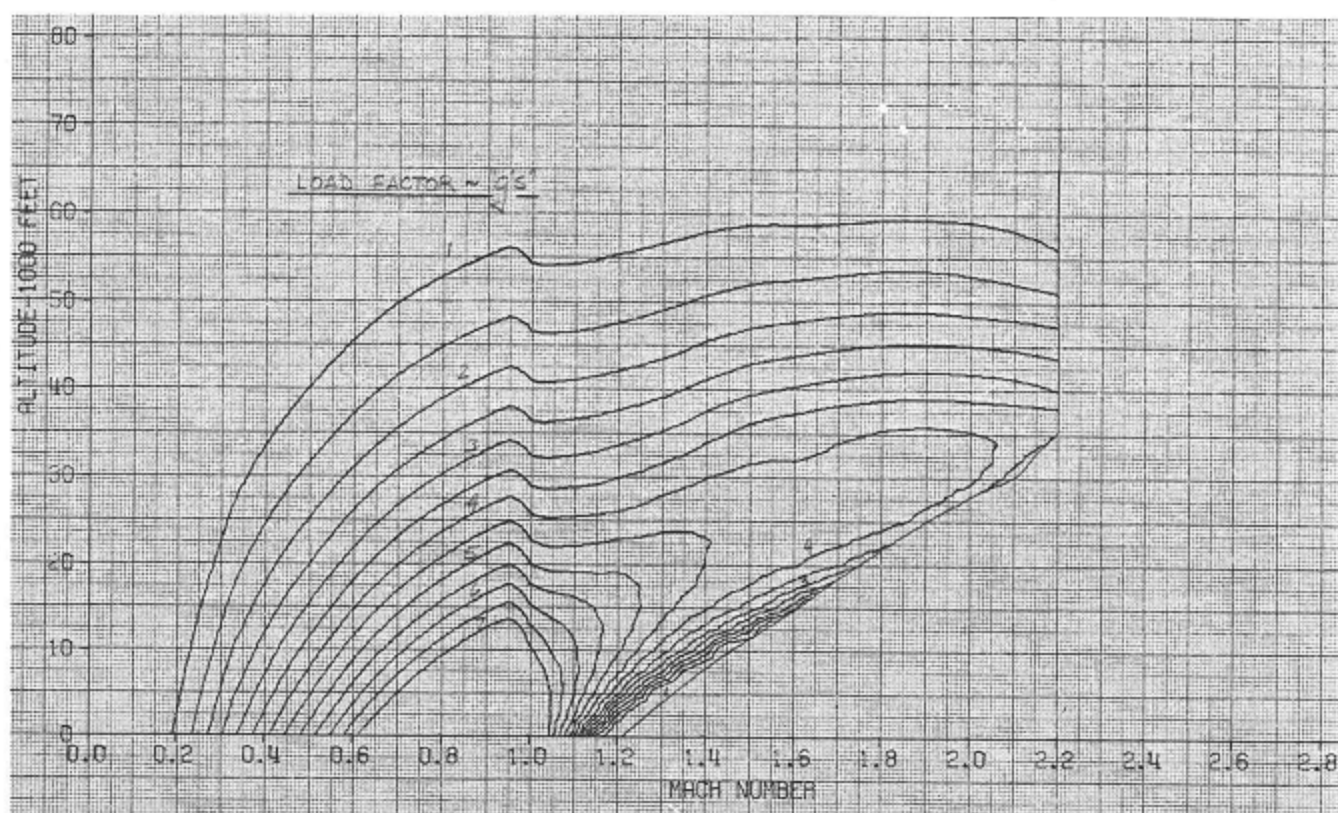


Figure 2-43. Steady State Load Factor, Clean Airplane

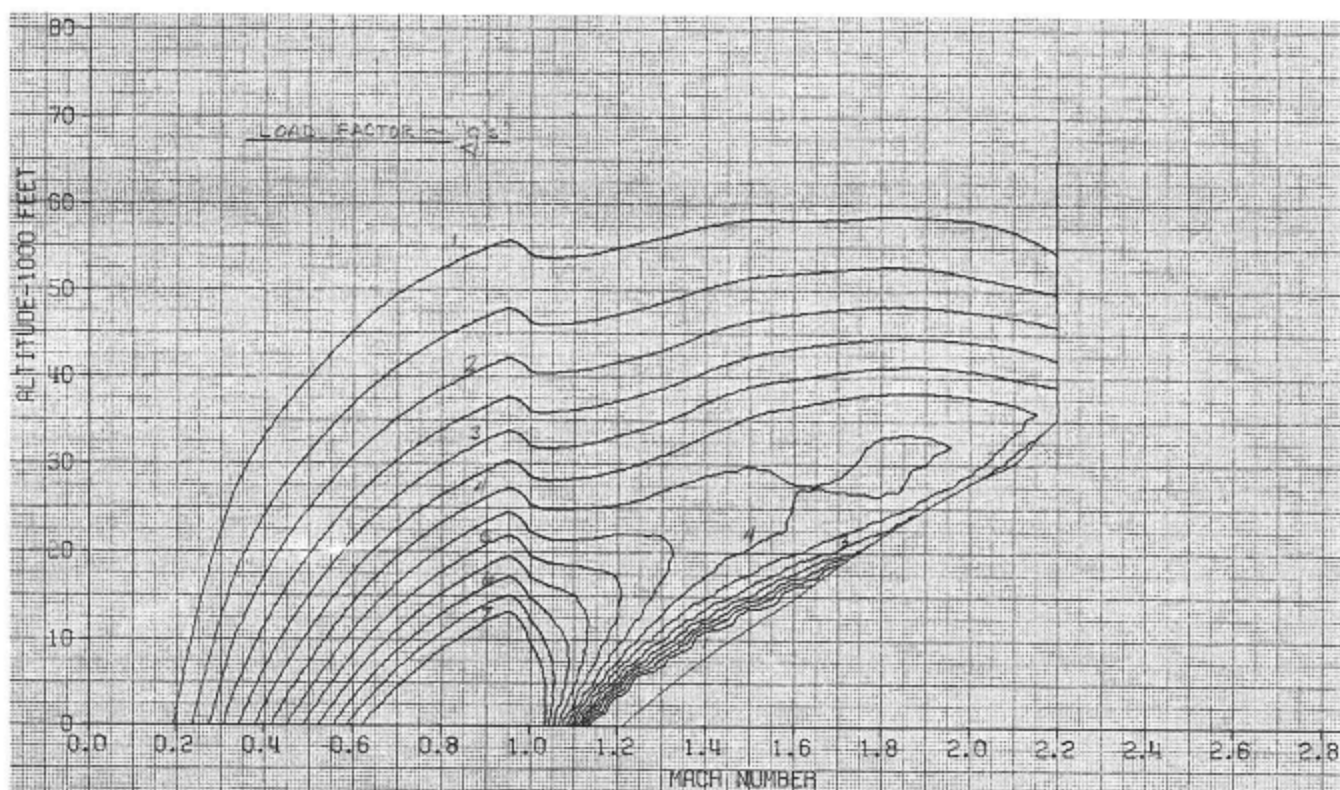


Figure 2-44. Steady State Load Factor, (2) Tip AIM-9

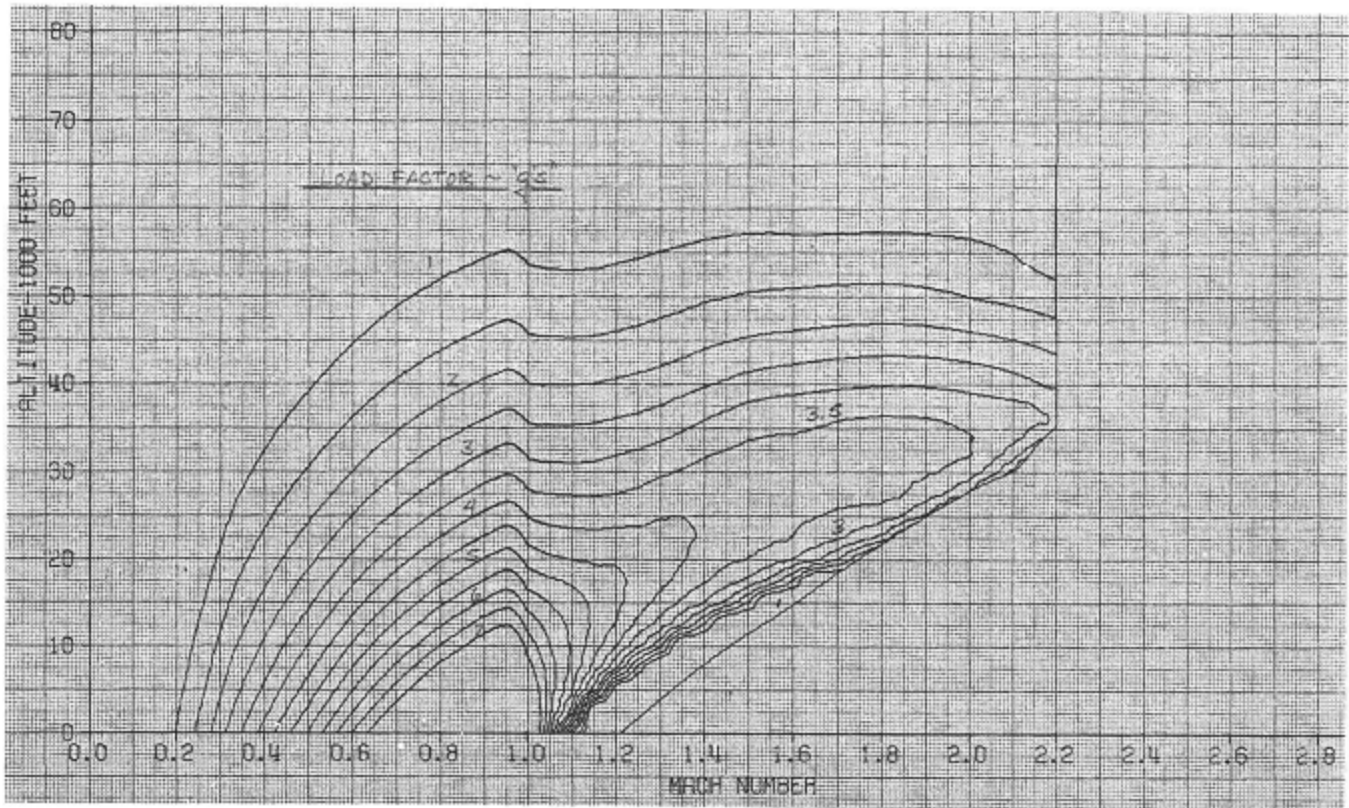


Figure 2-45. Steady State Load Factor, (2) Tip AIM-7

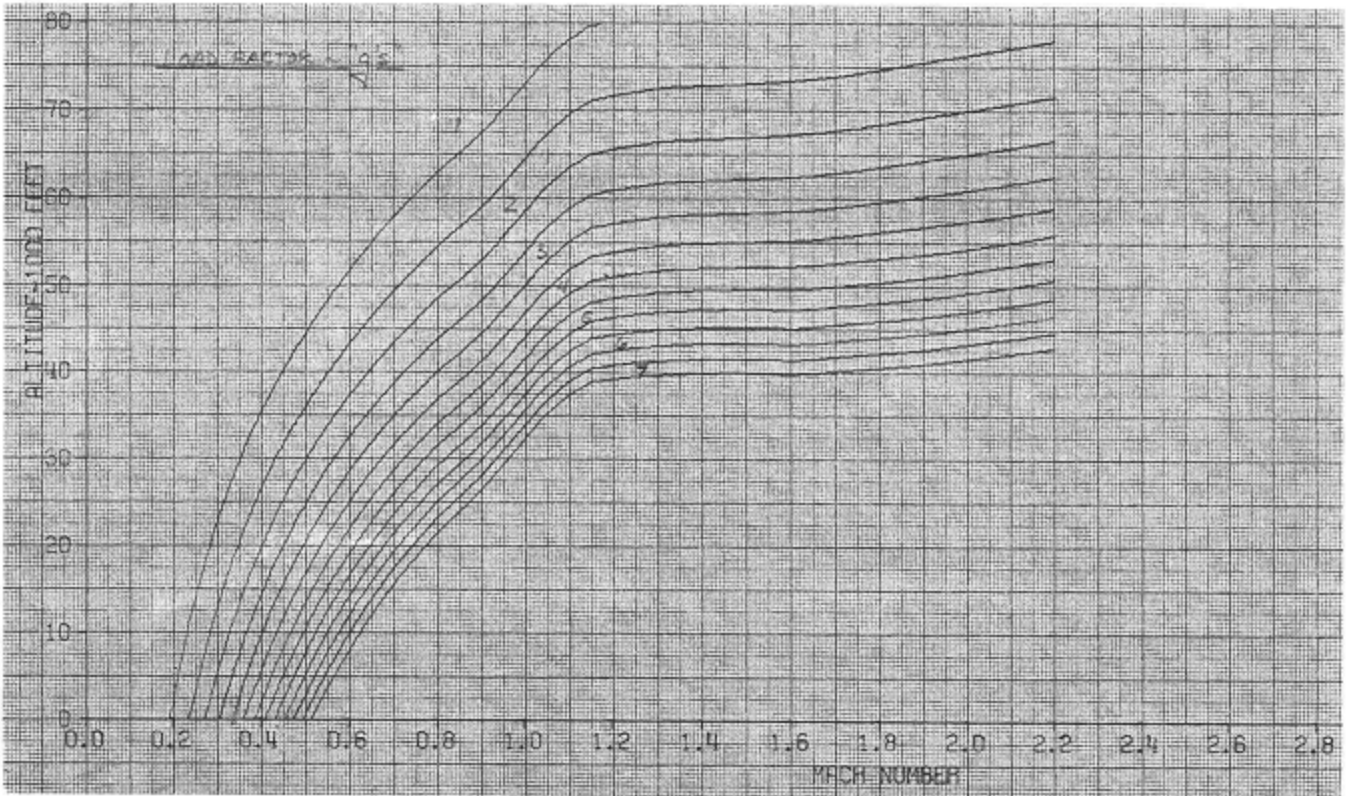


Figure 2-46. Max Instantaneous Load Factor, Clean Airplane

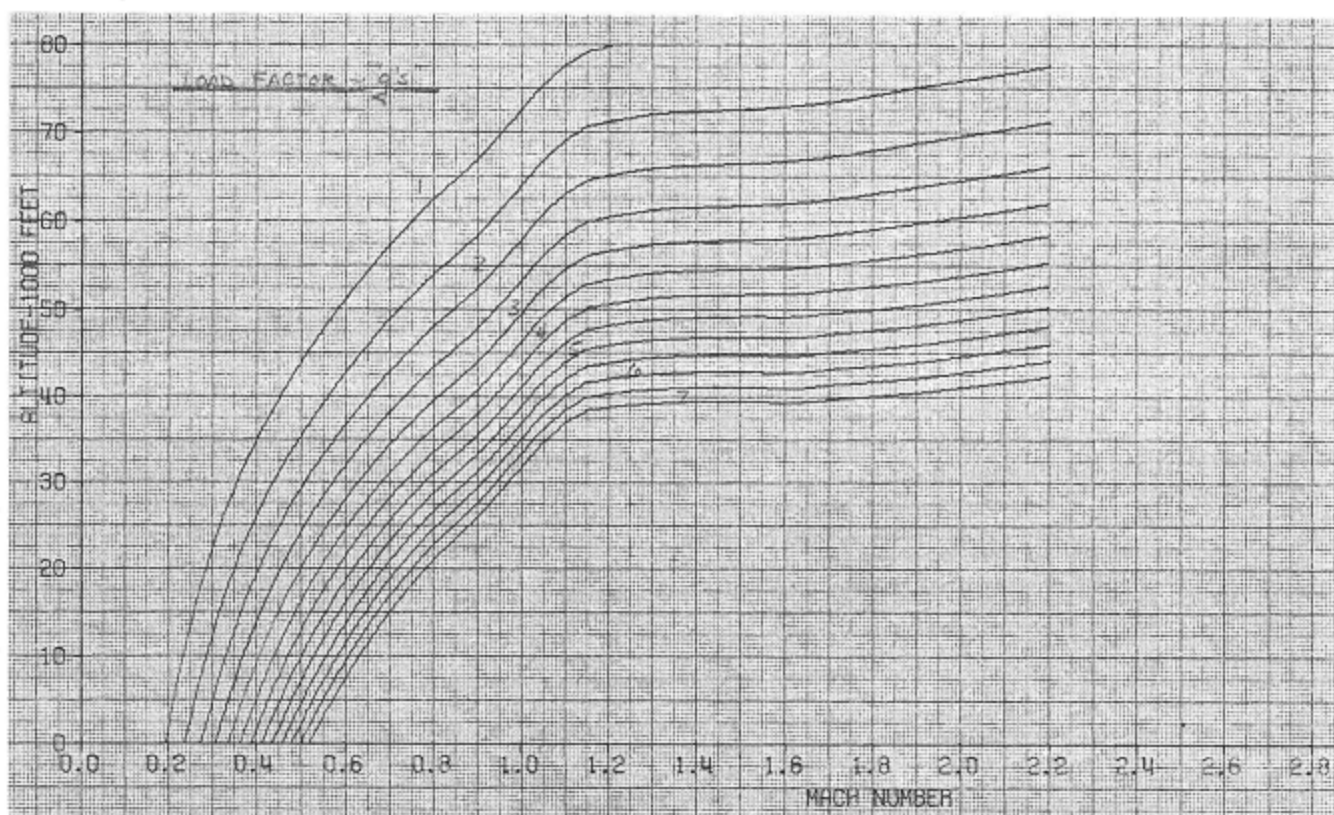


Figure 2-47. Max Instantaneous Load Factor, (2) Tip AIM-9

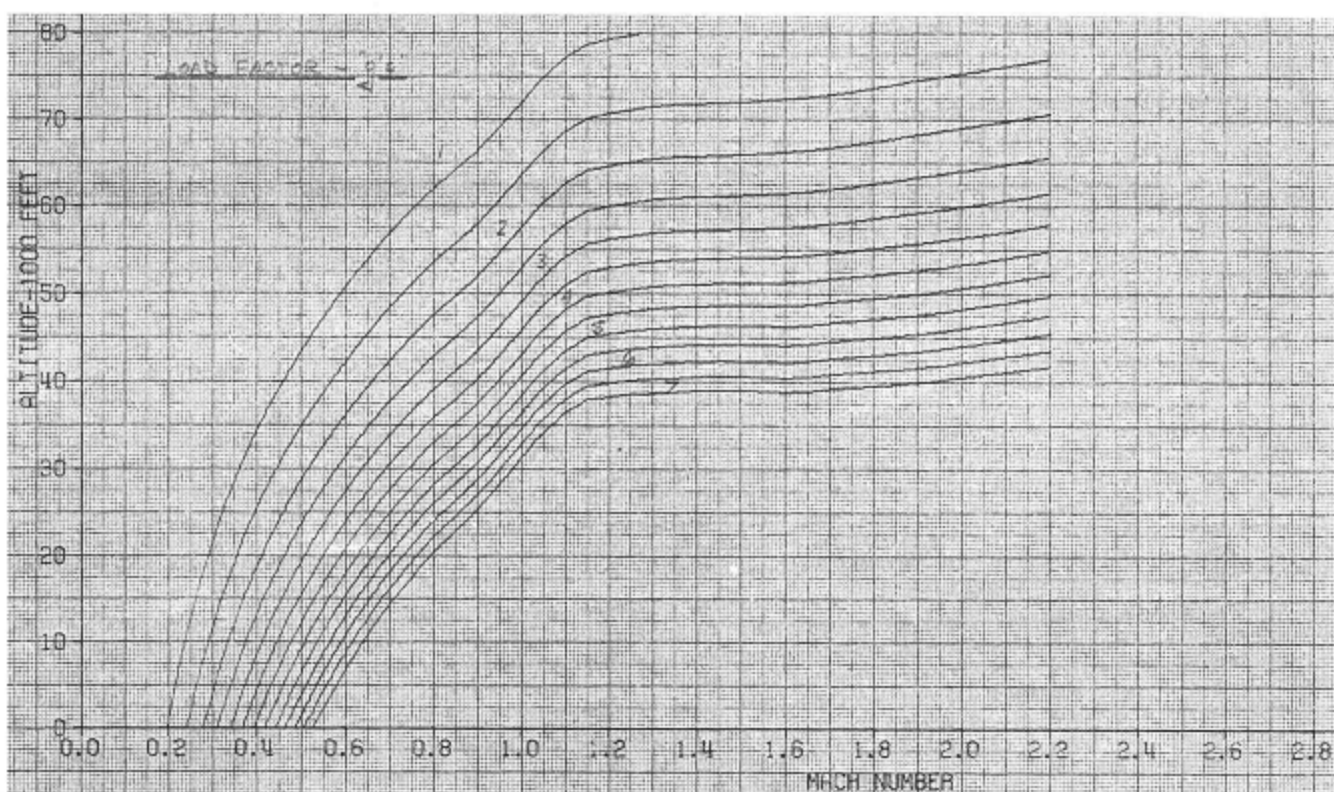


Figure 2-48. Max Instantaneous Load Factor, (2) Tip AIM-7

2.8 TURN RATE CAPABILITY

Steady state and maximum instantaneous level turn rates are presented versus Mach number in Figures 2-49 through 2-51, for 5,000, 15,000 and 30,000 foot altitudes. These turn rates correspond to the sustained and instantaneous load factors at constant altitude shown in the load factor capability plots of Figures 2-43 through 2-48.

The variation of SEP with level turn rate is shown in Figures 2-52 through 2-58, for altitudes from sea level to 30,000 feet, at 0.9 Mach number and (at most altitudes) one supersonic Mach number.

2.9 DESIGN LIMIT LOAD FACTORS

The Lancer is designed to a structural limit load factor of plus 6.5 g at the Design Combat Weight of 22,236 pounds. Load factor capability at other weights is based on a constant nW product to a maximum of 7.5 g at light weights. Limit load factor therefore varies with the weight of each of the Lancer configurations shown in the combat performance plots. For this reason, performance is shown up to 7 g in these plots, for interpolation to the actual limit load factor. The limits are shown for each configuration in Figure 2-49. These limits also apply to the SEP and load factor plots of Figures 2-22 through 2-48.

2.10 EFFECT OF COMBAT WEIGHT ON PERFORMANCE

All of the Lancer combat performance is presented at the traditional half-internal fuel weight. This is 4538 pounds of fuel for the Lancer and results in a high combat weight in all configurations. The Lancer's high internal fuel fraction achieves good mission capability, but penalizes combat performance in traditional comparisons. For this reason, Figure 2-59 is provided. It shows the effect of weight on sustained combat performance and level flight SEP. It is Lockheed's view that the performance of different fighter aircraft must be compared at a specified radius or for a specified combat time and hence fuel load. If the Turkish Air Force Foundation desires to define any specific mission criteria or evaluation point, Lockheed will promptly provide complete performance data for the Lancer.

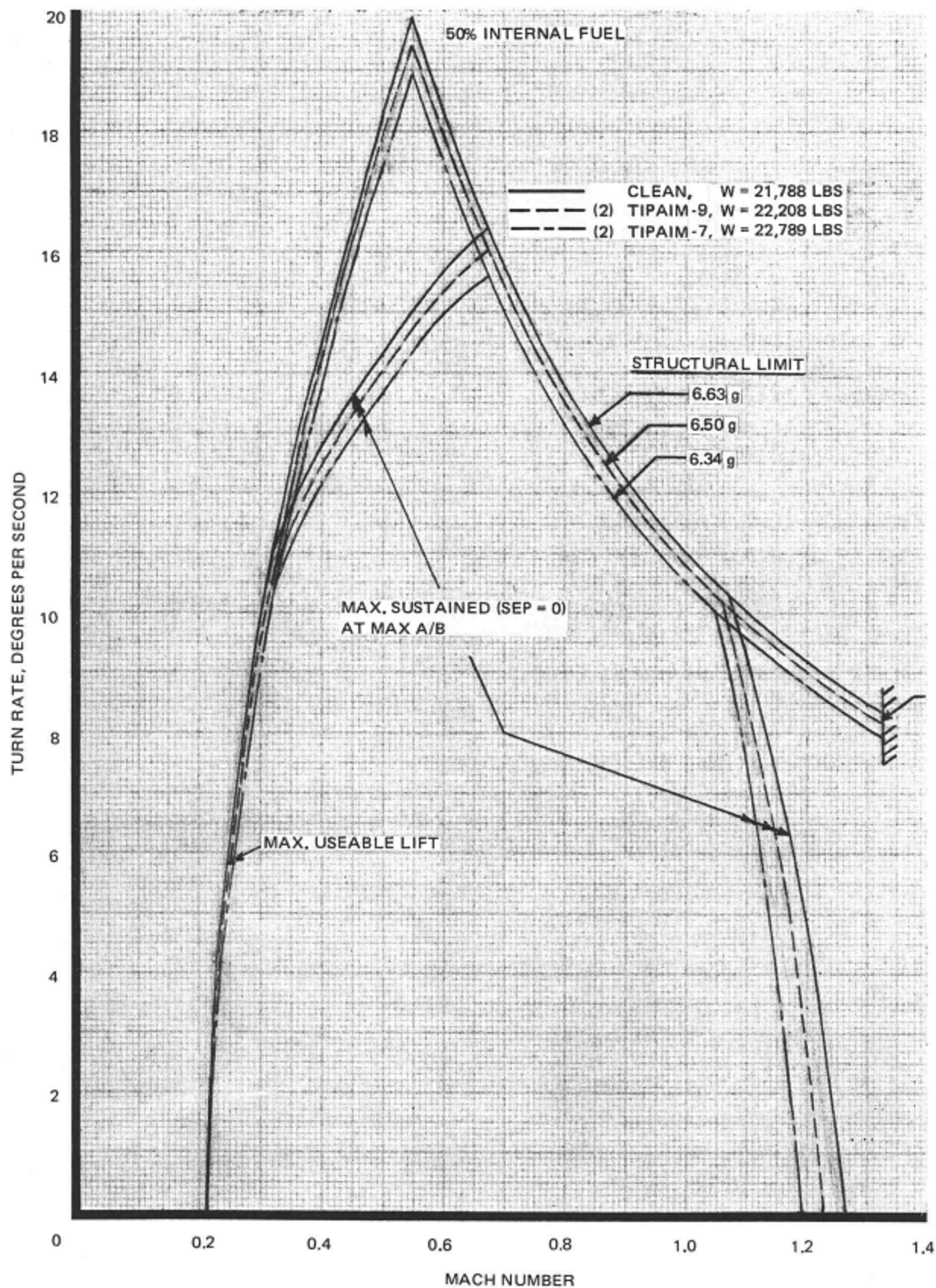


Figure 2-49. Turn Rate Vs. Mach Number, 5000 Ft. Alt.

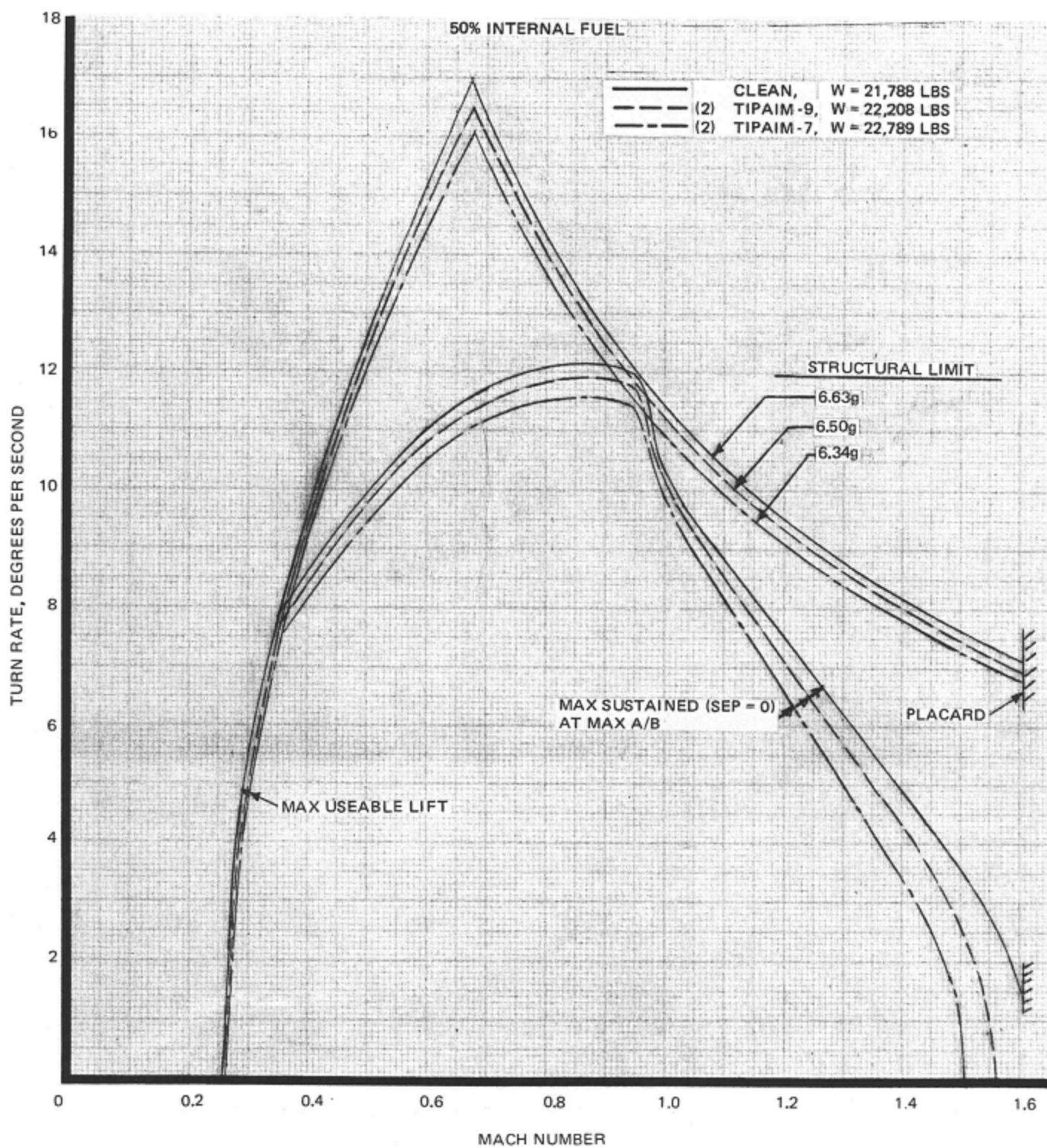


Figure 2-50. Turn Rate Vs. Mach Number, 15,000 Ft. Alt.

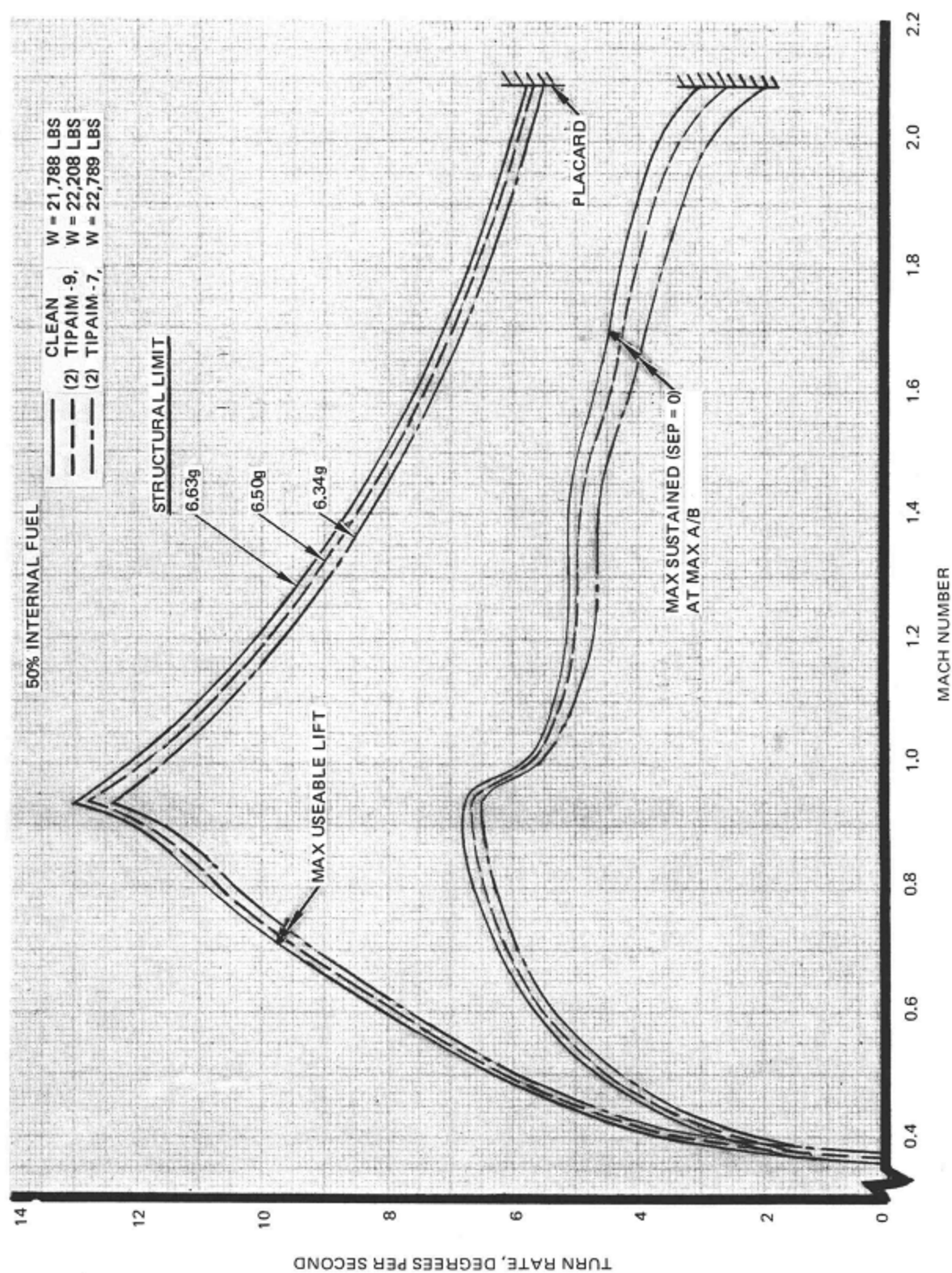


Figure 2-51. Turn Rate Vs. Mach Number, 30,000 Ft. Alt.

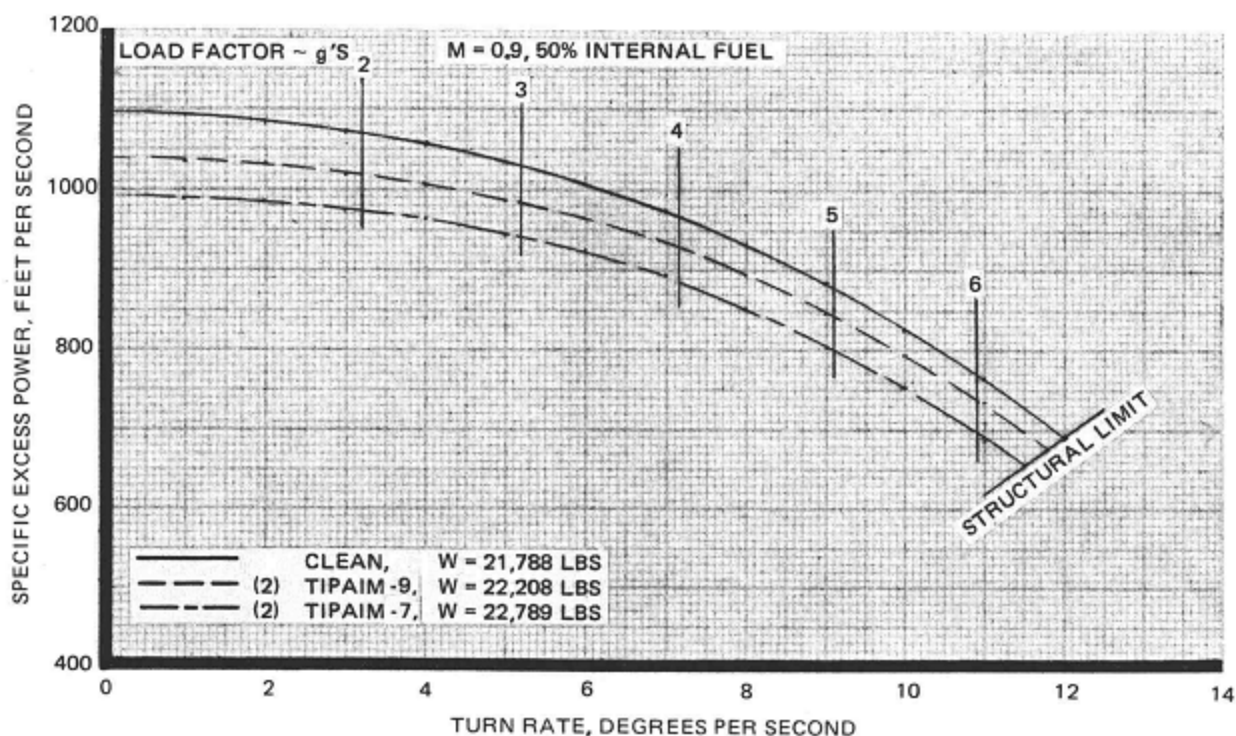


Figure 2-52. SEP Vs. Turn Rate, Sea Level

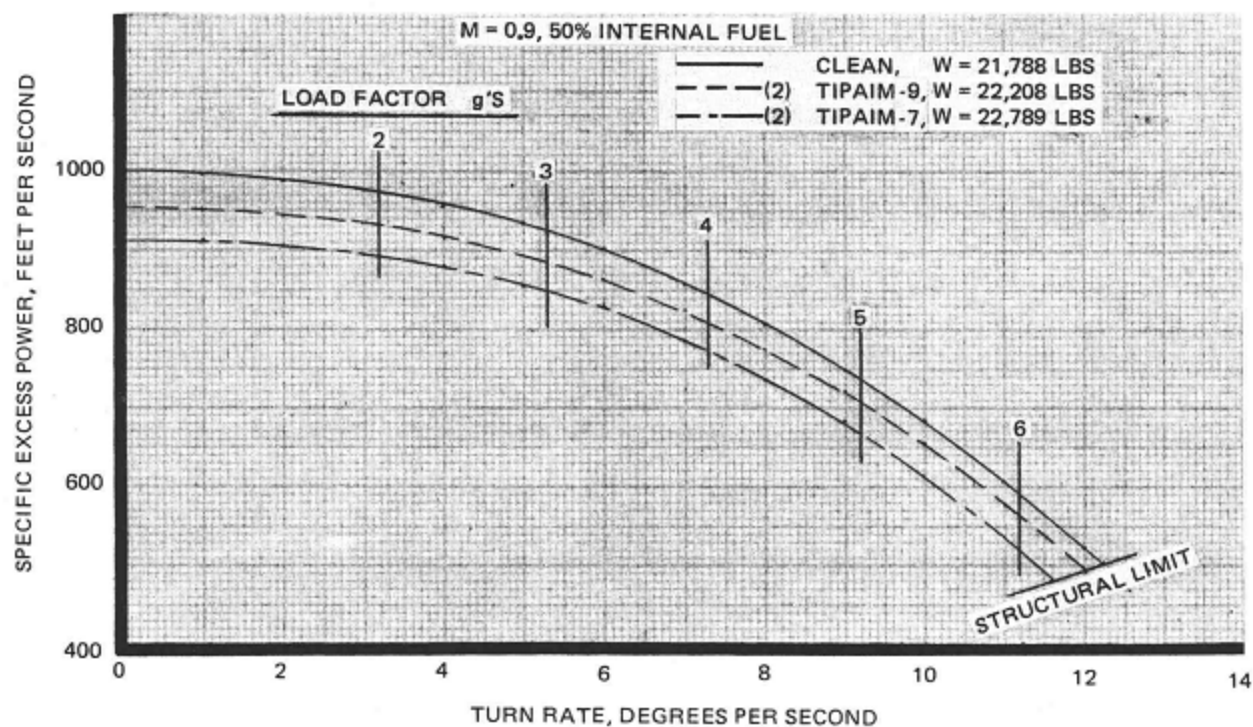


Figure 2-53. SEP Vs. Turn Rate, 5,000 Ft. Alt.

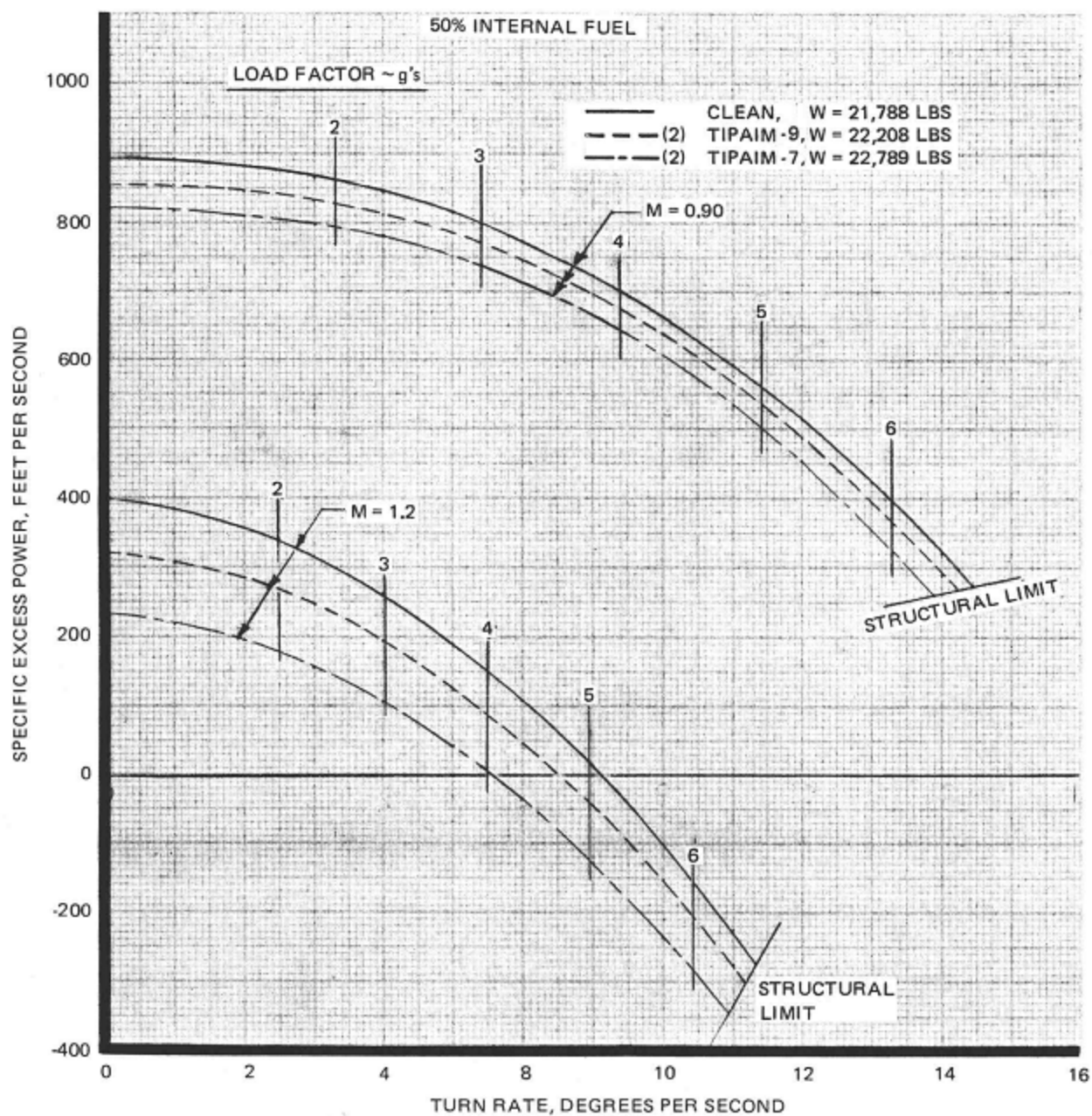


Figure 2-54. SEP Vs. Turn Rate, 10,000 Ft. Alt.

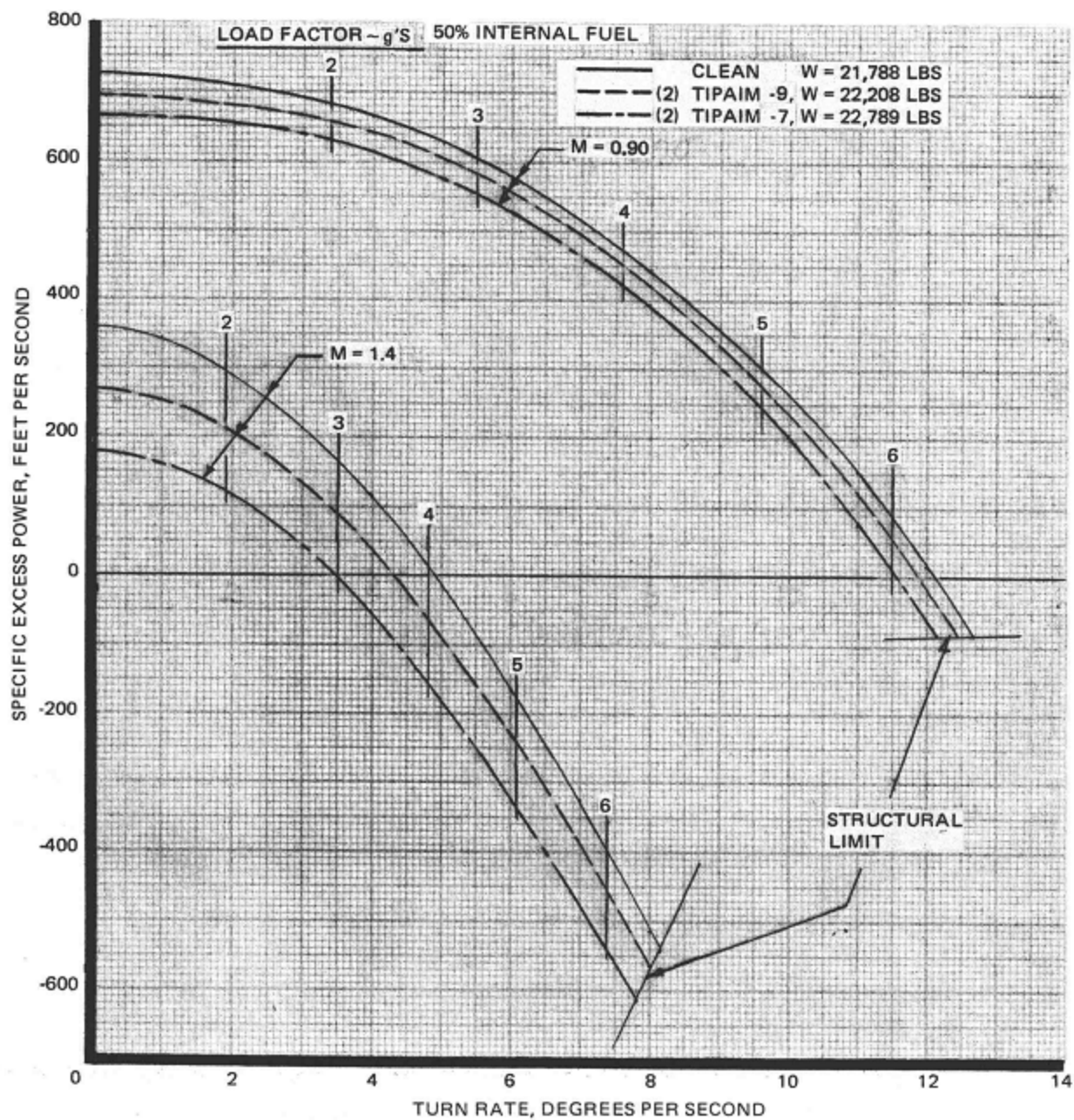


Figure 2-55. SEP Vs. Turn Rate, 15,000 Ft. Alt.

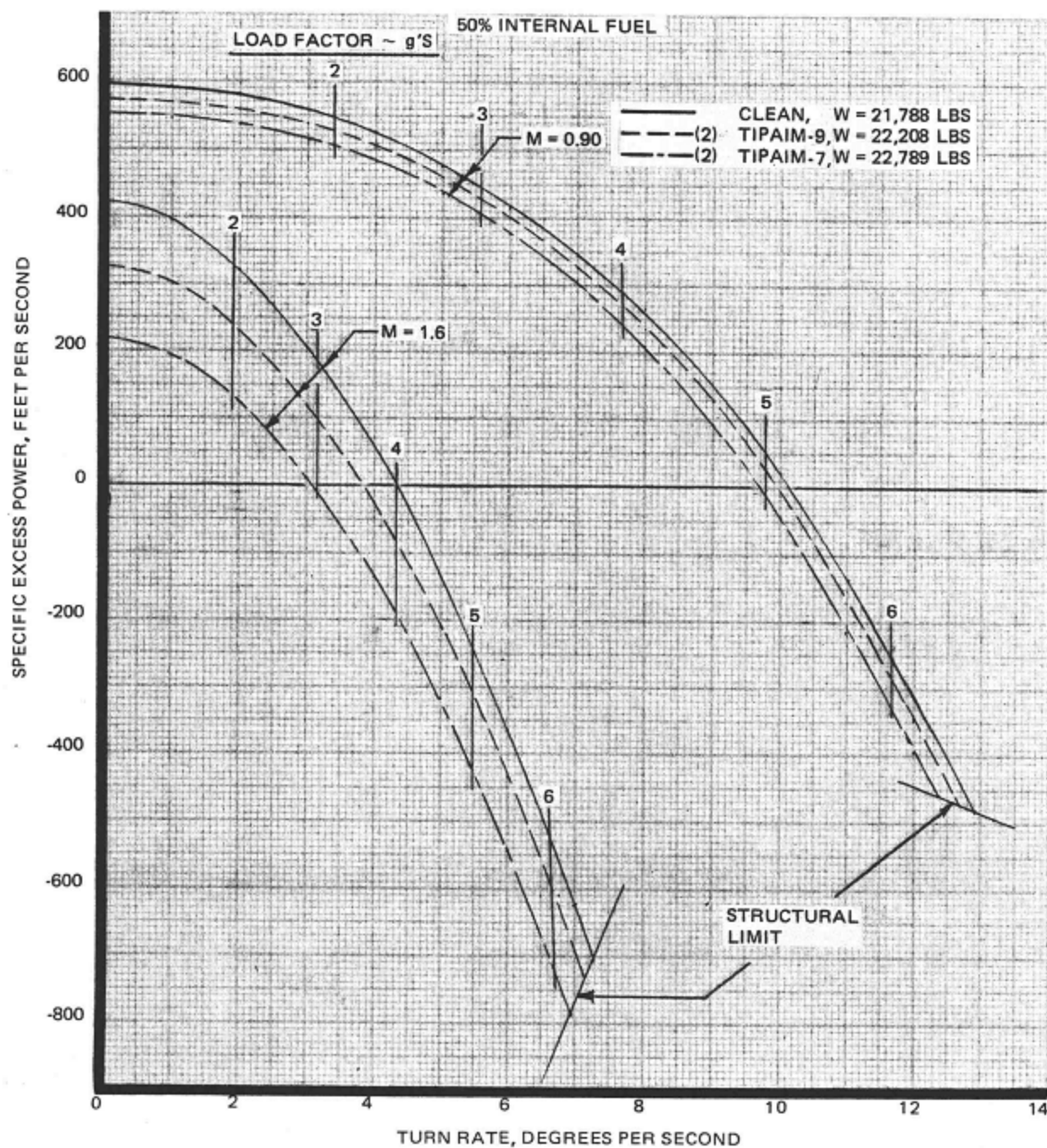


Figure 2-56. SEP Vs. Turn Rate, 20,000 Ft. Alt.

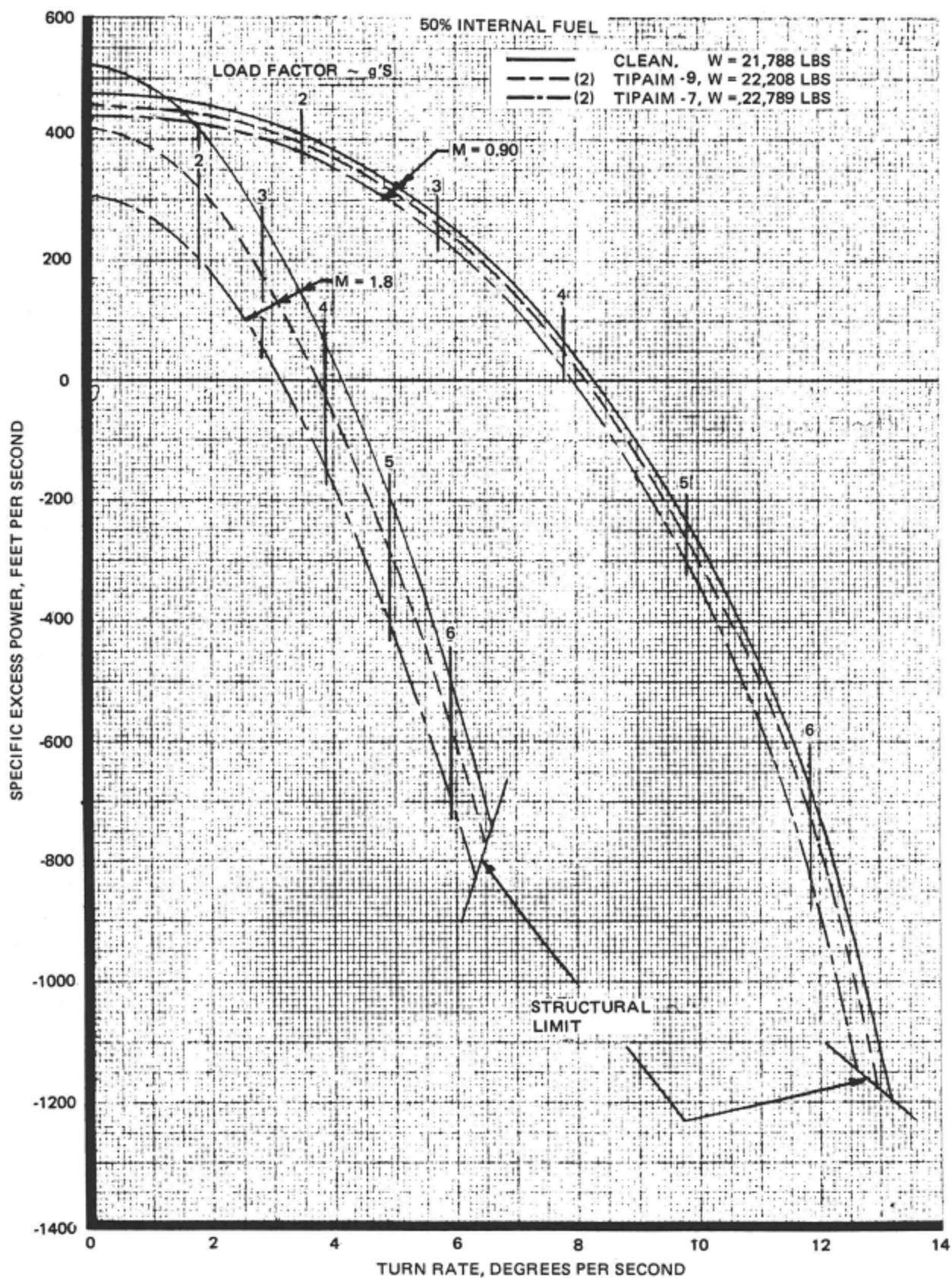


Figure 2-57. SEP Vs. Turn Rate, 25,000 Ft. Alt.

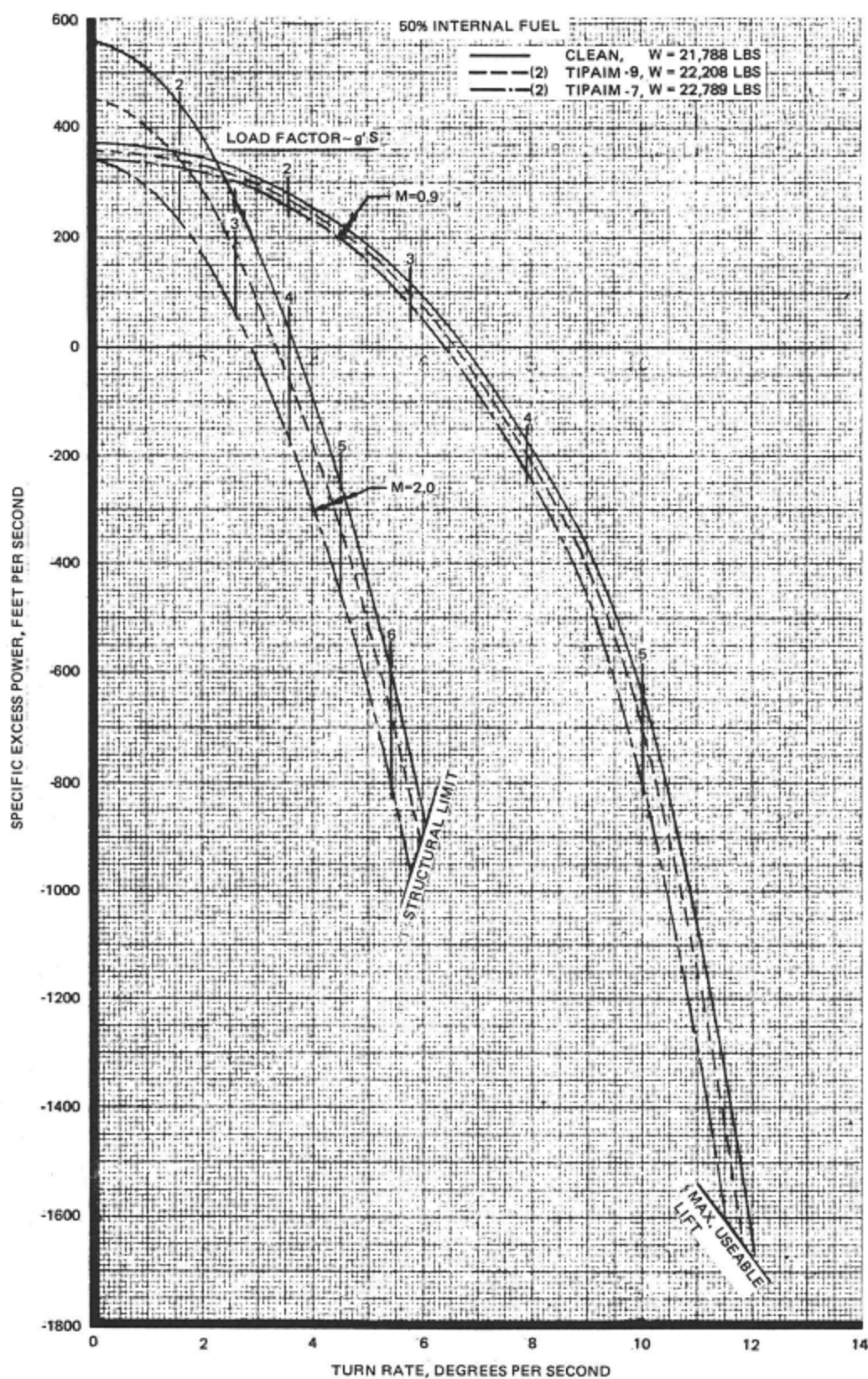


Figure 2-58. SEP Vs. Turn Rate, 30,000 Ft. Alt.

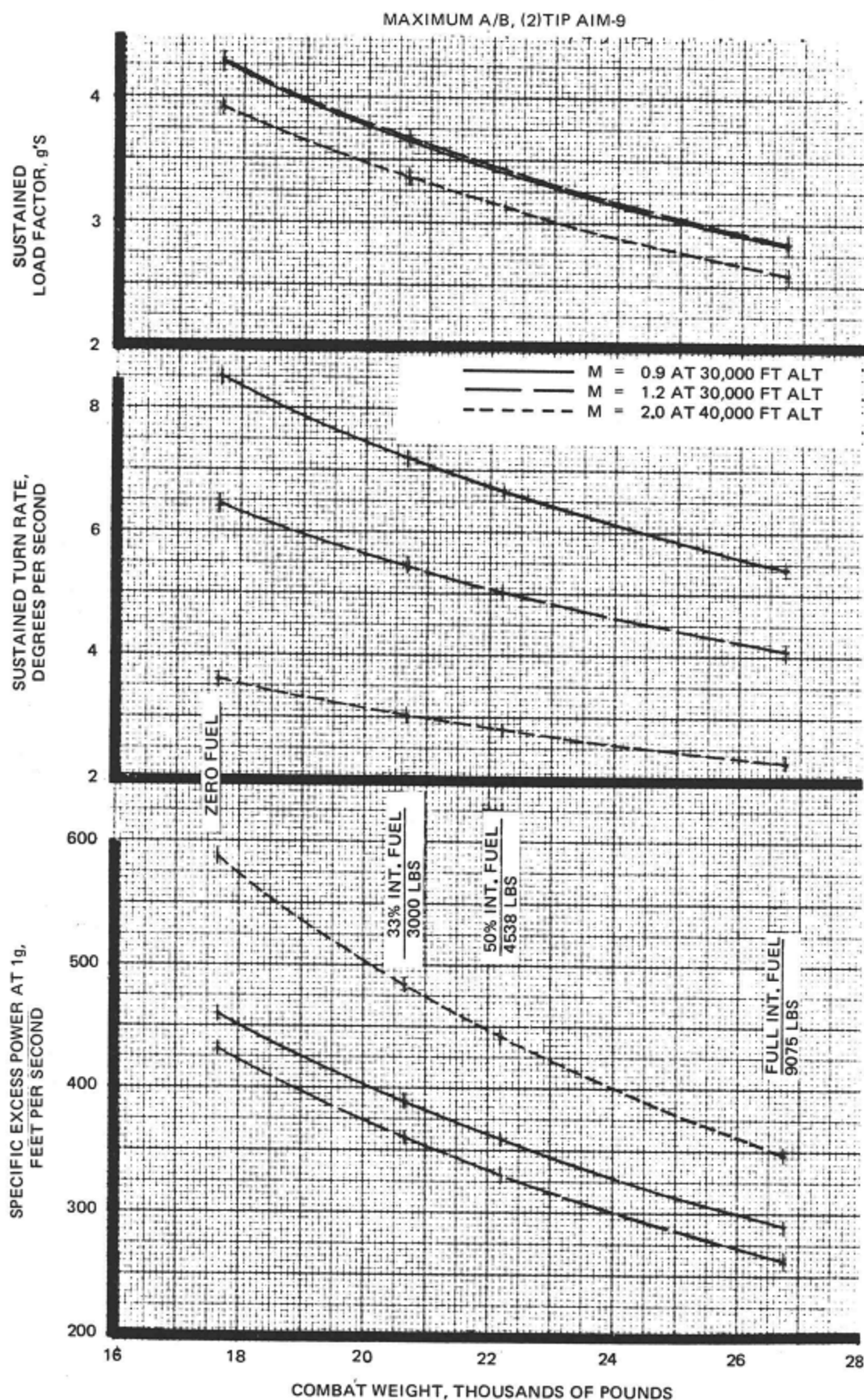


Figure 2-59. Effect of Weight on Combat Maneuverability

SECTION 3

STABILITY, CONTROL, AND HANDLING QUALITIES

The Lancer is designed to meet stringent requirements for maximum agility and good handling qualities in combat, since these characteristics are equally as important as performance in a fighter airplane. During the configuration development, therefore, a large portion of the wind tunnel test hours were devoted to determining basic stability and control characteristics over the full range of Mach numbers and angles of attack. The tests were conducted both with and without flowing inlets to assess the effects of the engine-air induction system on the stability of the airplane. Alternate types and sizes of pitch, roll, and yaw controls have been tested and evaluated in detailed stability and control analyses. These analyses included the effects of structural aeroelasticity over the Mach number and dynamic pressure range of the performance envelope. Concurrent control system selection and design studies have resulted in the final Lancer configuration proposed to the Turkish Air Force Foundation.

3.1 AIRCRAFT CONFIGURATION

The basic Lancer configuration provides high levels of aerodynamic stability throughout the Mach number and angle of attack range. The high-wing, low-tail arrangement produces increasing pitch stability with increasing angle of attack. The large, all-movable vertical tail provides high levels of directional stability and control, at all Mach numbers, for inherent spin-resistance. The Lancer's wing and horizontal stabilizer anhedral angles of 10 degrees each have been selected from dynamic lateral-directional analyses for excellent Dutch roll characteristics, utilizing wind tunnel data for 0, 5, and 10 degree wing anhedrals. The 10 degree wing anhedral is similar to the F-104's and permits the use of the Lancer external store wing pylons without redesign.

3.2 BASIC FLIGHT CONTROLS

The Lancer's high inherent stability margins result in an airplane which forgives inadvertent overcontrol, but dictate high required control power if agility is to be maintained. All-movable horizontal and vertical stabilizers have been selected to develop the desired level of control power. Large, plane flap outboard ailerons are provided for roll control. Roll spoilers are not utilized due to their relative inefficiency at high angles of attack and in spin recovery, as verified in the wind tunnel development tests. Differential stabilizer deflection for roll control



was also tested and is not used because of added system complexity, the associated large aerodynamic roll-to-yaw coupling, and the loss of pitch control when combining functions at large deflections.

The Lancer's all-movable vertical tail provides large sideslip angle and lateral acceleration capability, a distinct advantage in bringing the gun to bear in close, turning combat. It also provides the 30-knot crosswind capability in landings and takeoffs required by the latest U.S. MIL-SPEC and is very effective in spin recovery.

3.3 AUTOMATIC FLIGHT CONTROL SYSTEM AND HANDLING QUALITIES

The Lancer's high levels of aerodynamic stability and control power are integrated by the Automatic Flight Control System (AFCS) to provide superior aircraft handling qualities throughout the flight envelope. The AFCS augments damping about all three axes, static pitch stability as a function of lift coefficient, and static directional stability as a function of both lateral and yaw accelerations. The AFCS tailors the aircraft responses to provide linear stick forces per g in pitch and roll rate proportional to stick force in roll. The system provides increased stick force per g near the limit load factor as a cue to prevent overstressing the airplane. The Mach trim function of the AFCS provides stable stick forces versus speed and stabilizes the airplane phugoid motion.

In selecting and designing the Lancer control systems, the skills of automatic flight control specialists experienced on the Lockheed ADP SR-71 Mach 3.0+ aircraft have been utilized. The opinions of test pilots with actual combat experience have been solicited and embodied in the selection of control forces, gradients and functions.

3.4 ROLL PERFORMANCE

The maximum roll performance of the Lancer meets the stringent U.S. MIL-SPEC combat requirements for fighter aircraft, throughout the subsonic and transonic Mach number range, for both 1.0 g and 5.0 g load factors. Roll-coupling analyses have also been conducted and show that no unstable inertial coupling conditions exist within the structural design limits of the airplane, for maximum deflection aileron rolls.

3.5 SPIN RESISTANCE

The Lancer has been designed for high inherent spin resistance. This is provided by high static and dynamic directional stability throughout the Mach number and angle of attack range. The "lateral control spin parameter" has also been evaluated

to define the maximum usable angle of attack upon which all performance lift limits (maximum lift) are based. The lateral control spin parameter relates lateral control power, sideslip due to lateral control, directional stability, and the inertial characteristics of the airplane to ensure no adverse "roll-off" tendencies at high angles of attack. This parameter, the dynamic directional stability derivative, and the pitching moment due to sideslip from the wind tunnel tests are indicative of a low susceptibility to spin at high angles of attack. The AFCS will also increase the spin resistance through its influence on damping, stability, and stick force. The high directional effectiveness of the all-movable vertical stabilizer will provide positive, normal recovery from fully-developed intentional spins.

SECTION 4
STRUCTURAL AND SYSTEMS DESCRIPTIONS

4.1 AIRFRAME DESCRIPTION, STRUCTURES AND WEIGHTS

This subsection describes the structural and weight and balance characteristics of the Lancer. Included are a brief description of the airframe structure, design criteria, structural materials, aircraft surface aeroelasticity effects, and weight and balance data.

REFERENCES

Specifications:

MIL-A-8860	Airplane Strength and Rigidity-General (18 May 1960)
MIL-A-8861	Airplane Strength and Rigidity-Flight Loads (18 May 1960)
MIL-A-8862	Airplane Strength and Rigidity-Ground Loads (18 May 1960)
MIL-A-8865	Airplane Strength and Rigidity-Miscellaneous Loads (18 May 1960)
MIL-A-008866A	Airplane Strength and Rigidity-Reliability, Repeated Loads and Fatigue (31 March 1971)
MIL-A-8870	Airplane Strength and Rigidity-Vibration, Flutter and Divergence (18 May 1960)
MIL-W-5013G	Wheel and Brake Assemblies, Aircraft (20 Feb. 1967)
MIL-T-5041F	Tires, Pneumatic, Aircraft (26 March 1971)

Manuals:

Lockheed Structural Life - Assurance Manual

4.1.1 Airframe Description

The Lancer airplane's general arrangement and basic dimensions are shown in Figure 4-1. The main elements of the airframe structure are shown in the inboard profile drawing, Figure 4-2.

4.1.1.1 Fuselage. The fuselage structure is of semi-monocoque construction utilizing multiple longerons, skin, frames, and bulkheads. The cockpit structure, canopy and windshield are a pressurized compartment. The forward fuselage, up to Fuselage Station (F.S.) 358, utilizes all the major structural components of the F-104S. The total intent of the design is to capitalize and make use of any major F-104S components where feasible. Engine air inlets are pressure designed shells consisting



of skin and multiple frames. The engine air inlet outer shell supports the fixed forward wing deltas. The wing-to-fuselage attachment consists of multiple links and fittings (six on each side) attached to the six fuselage main frames for transmission of vertical shear and torsion loads. Drag truss links are located on each side, and side links at front and rear spars. This arrangement is fail-safe and also prevents undesirable wing-fuselage deflection interaction. Landing gear supports are similar to the proven F-104S design. The engine is supported on forward and aft engine mount frames. The all-movable tail surfaces, both vertical and horizontal, are pivoted on posts which are designed for safe-life. The posts are supported in the fuselage by a set of two fail-safe designed frames. Fail safe dictates that the structure, though damaged to a limited extent, is capable of sustaining a reasonable percentage of load, and safe operation of the aircraft is not impaired. A production break occurs at F.S. 358 and a service and maintenance break at F.S. 646. The latter also provides for engine removal.

4.1.1.2 Wing. The wing consists of a main box structure, leading and trailing edge flaps, ailerons and fixed forward deltas. The main box extends over the fuselage and has a fail-safe structural design using multiple spars and machine tapered skins divided into four panels. Upper surface panels are removable from the centerline to Wing Station 112 for access to the integral fuel tank. Major ribs are located at the centerline joint, fuselage attachment, pylon stations, and wing tips. The wing attachment is described in paragraph 4.1.1.1. The leading edge flaps are divided into three sections; the ailerons and trailing edge flaps are each divided into two sections and each section is supported by two hinges. This design minimizes the effects of wing bending deflection on the control surfaces. The wing planform is similar to the F-104S. The pylon and tip stations are designed to accommodate all types of F-104S external stores, fuel tanks and ordnance.

4.1.1.3 Vertical Stabilizer. The large, upper portion of the vertical stabilizer is all-movable. It is mounted on a pivot post supported in the aft fuselage by extensions of the paired, fail-safe designed empennage support frames. The frame extensions also support the stub fin which contains the split trailing-edge speedbrake and the servo system for both the vertical stabilizer and speedbrake. The speedbrake was moved from the fuselage to preclude interference with the horizontal tail and to increase the surface area. The stabilizer structural box is constructed of multiple beams and multiple skin supporting ribs. Main ribs are located at the upper and lower bearings at the stabilizer root.

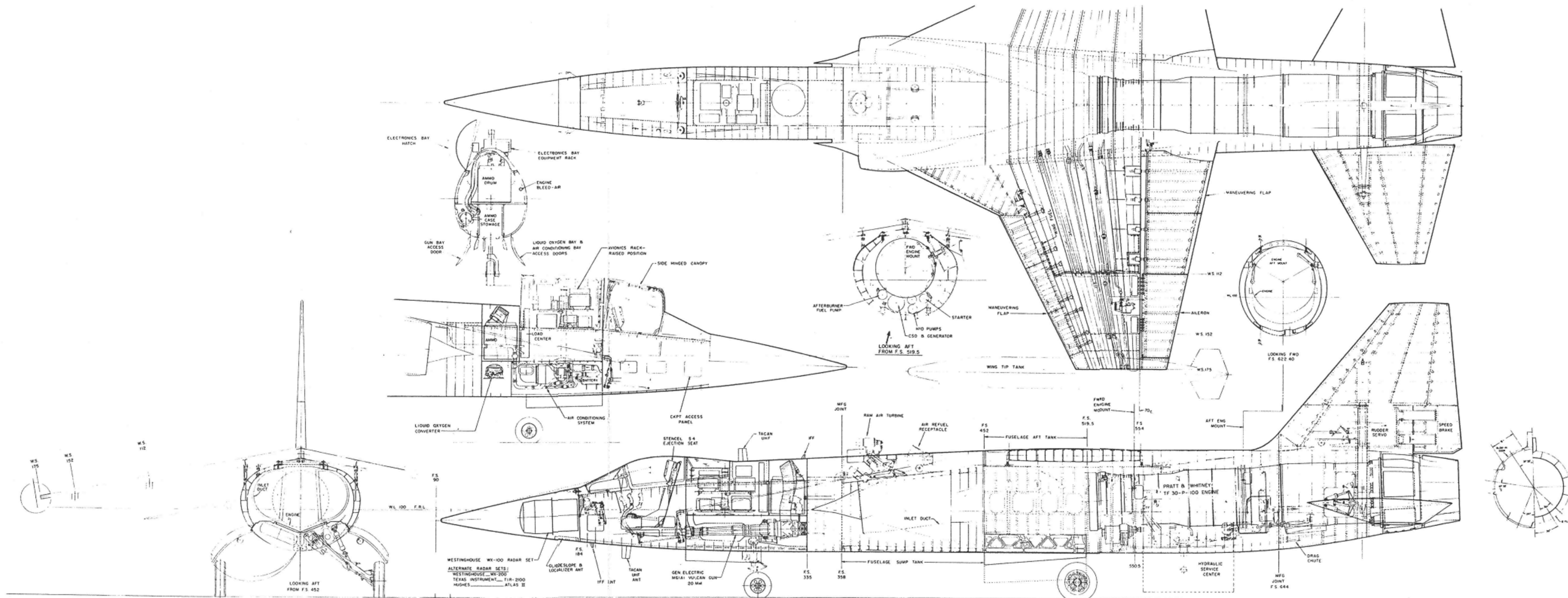


Figure 4-2. Inboard Profile, Lancer

4.1.1.4 Horizontal Stabilizer. All-movable horizontal stabilizers are mounted on pivot posts supported by the same pair of fuselage frames supporting the vertical stabilizer. Design of the horizontal surface is similar to that of the vertical, except that the root rib sweeps the fuselage side with no fixed stub surface. Actuators and servos are located inside the fuselage shell. External access is provided for servicing. The actuating mechanisms are interconnected to provide symmetrical deflection.

4.1.2 Structural Design Criteria

Structural design criteria for the Lancer are based on the general requirements of the MIL-A-8860 series of specifications, dated 18 May 1960, for a Class F_{II} aircraft. These general requirements, adapted to the specific mission requirements of the Lancer, result in the structural design criteria defined herein. These criteria also reflect Lockheed experience derived from worldwide operations of F-104 airplanes by fourteen nations.

4.1.2.1 Design Gross Weight. The Basic Design Combat Weight is based on the clean zero fuel weight of 17,250 pounds (including the M-61 gun and 500 rounds of ammunition) plus the AIM-9 advanced Sidewinder wingtip missile installations and 50 percent of full internal fuel (the full internal fuel load is 9075 pounds). The resulting Basic Design Combat Weight of 22,236 pounds at 6.5g is the baseline structural design point. Other design gross weights and the corresponding design limit load factors are summarized in Table 4-1. The 6.5g value was selected based on a careful study of Viet Nam combat experience.

4.1.2.2 Flight Loads Criteria. The speeds V_H , V_L , M_H and M_L , in addition to curves defining load factor and temperature and pressure limits from sea level to 60,000 feet, are shown by Figure 4-3. Mach number versus load factor diagrams for 5,000, 20,000, 35,000, and 55,000 feet are presented in Figure 4-4. These two figures are based on the Basic Design Combat Weight where the structural design limit load factors (n) are 6.5g positive and 2.6g negative. The variation in flight limit load factor versus gross weight, maintaining constant positive and negative products, is presented in Figure 4-5. Sidewinder missile installations are designed for full airplane load factors. Other external store installations are limited to 6g positive and 2g negative load factors or to the nW capability of the airplane up to a Mach number of 1.6. Sparrow missiles in the all-weather configuration can be carried to the limit Mach number of 2.2. Tip tanks and pylon tanks are limited to 5g and

TABLE 4-1. DESIGN WEIGHTS AND LIMIT LOAD FACTORS

DESCRIPTION	ARMAMENT			FUEL		LIMIT LOAD FACTOR		WEIGHT
	GUN	AMMO	STORES	INT	EXT	POS	NEG	
MIN FLYING WT	20 MM	EXPENDED		5%		7.5	3.0	17,704
LANDING DESIGN WT	20 MM	500 RNDs	①	①				22,000
BASIC DESIGN COMBAT WT	20 MM	500 RNDs	2 AIM 9's	50%		6.5	2.6	22,236
MAX TAKE-OFF WT	20 MM	500 RNDs	②	②	②	4.5	1.8	32,500 ③
NOTE ① EXTERNAL STORES AND INTERNAL FUEL MAY BE INTERCHANGED UP TO MAXIMUM STORE CAPACITY AND WITHIN LANDING DESIGN WEIGHT. ② SAME AS ①, WITHIN MAXIMUM TAKE-OFF WEIGHT. ③ ALSO MAXIMUM LANDING DESIGN WEIGHT.								

600 KEAS. The usual factor of safety of 1.5 is maintained between limit and ultimate loads.

4.1.2.3 Fatigue Criteria. New fatigue criteria for lightweight, high performance fighters have been developed for the Lancer, utilizing a new "normal" versus "combat" mode of operation and extensive worldwide F-104 experience. These criteria state that the airplane shall have a nominal required fatigue life of 4000 hours when operated to maneuver load factors not to exceed the specified limits. These limits are shown as dashed lines on Figures 4-4 and 4-5 for normal operations. This normal operations flight limit is based on 80 percent of the full structural design limit load factor. Furthermore, the airplane shall have a required life of 2000 hours when operated to full limit load factors during combat missions. The advantages of the operational plan and these criteria are that structural weight for fatigue is minimized, producing higher performance, and a long and economical fatigue life is achieved. Static strength requirements are met by retaining the usual ultimate safety factor of 1.5 times limit load.

The required fatigue life is achieved by making the proper choice of materials and processes, limiting design stress levels, and using fatigue resistant detail design.

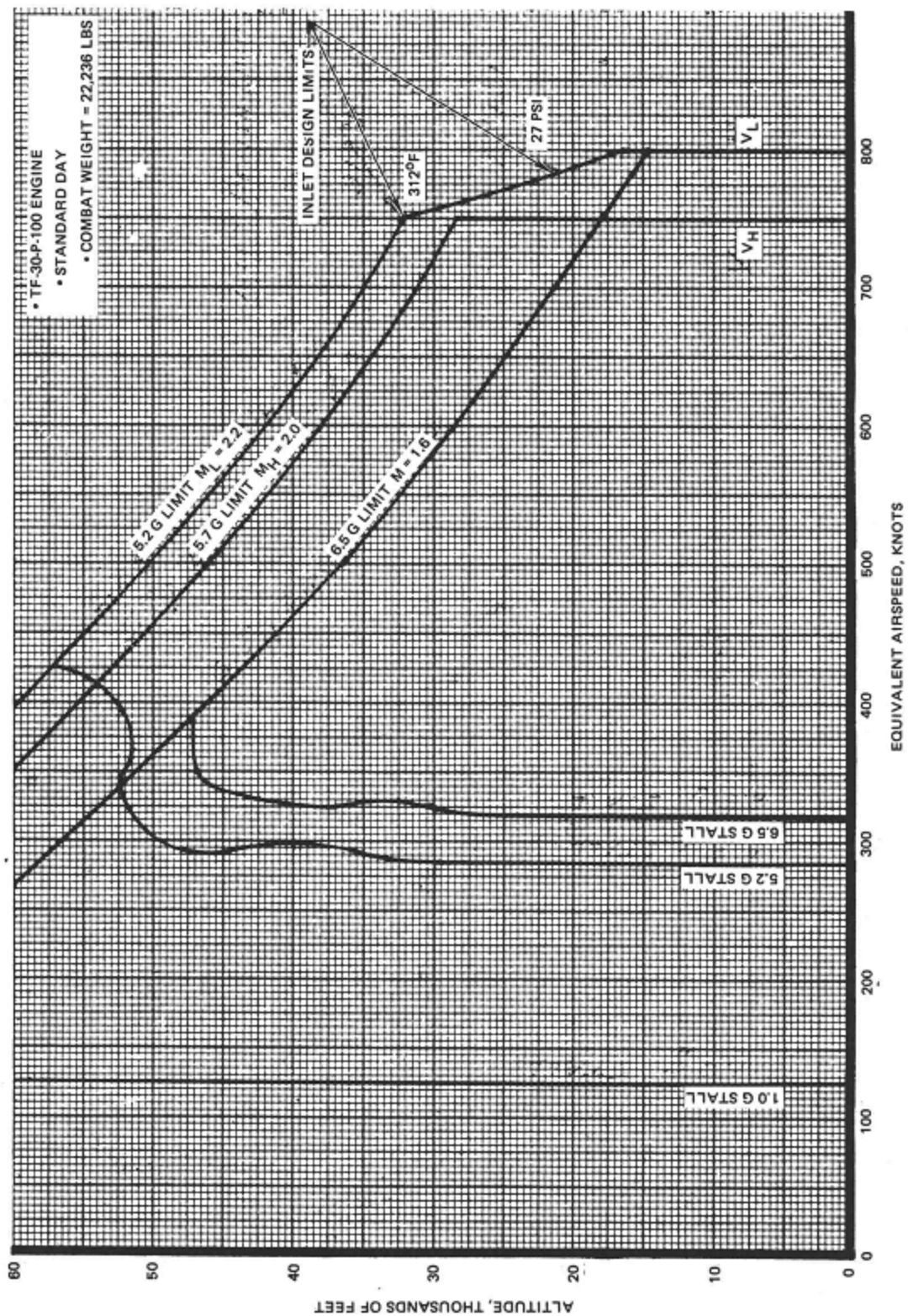


Figure 4-3. Structural Design Speed Envelope

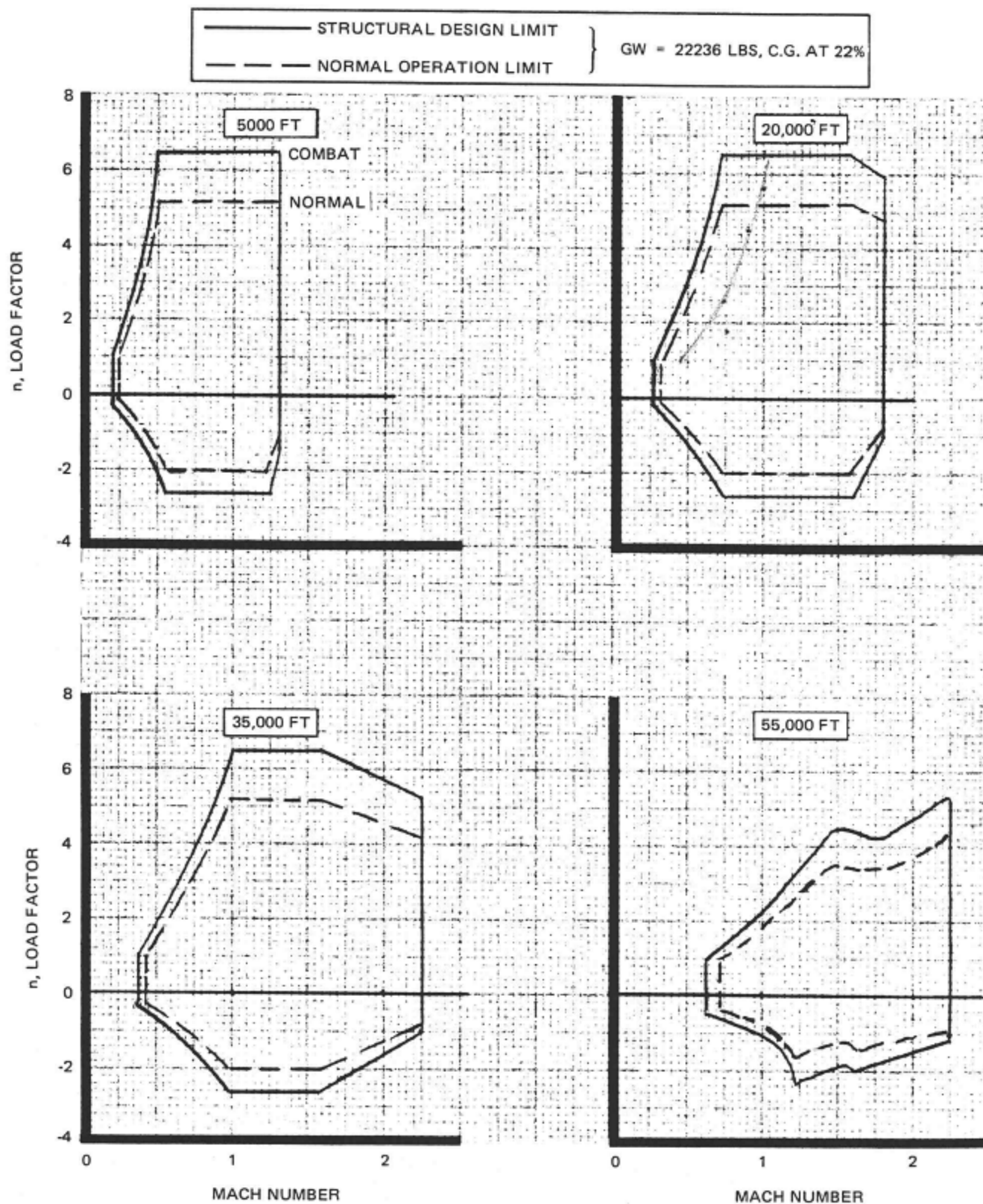


Figure 4-4. Mach Number Vs. Load Factor Diagrams

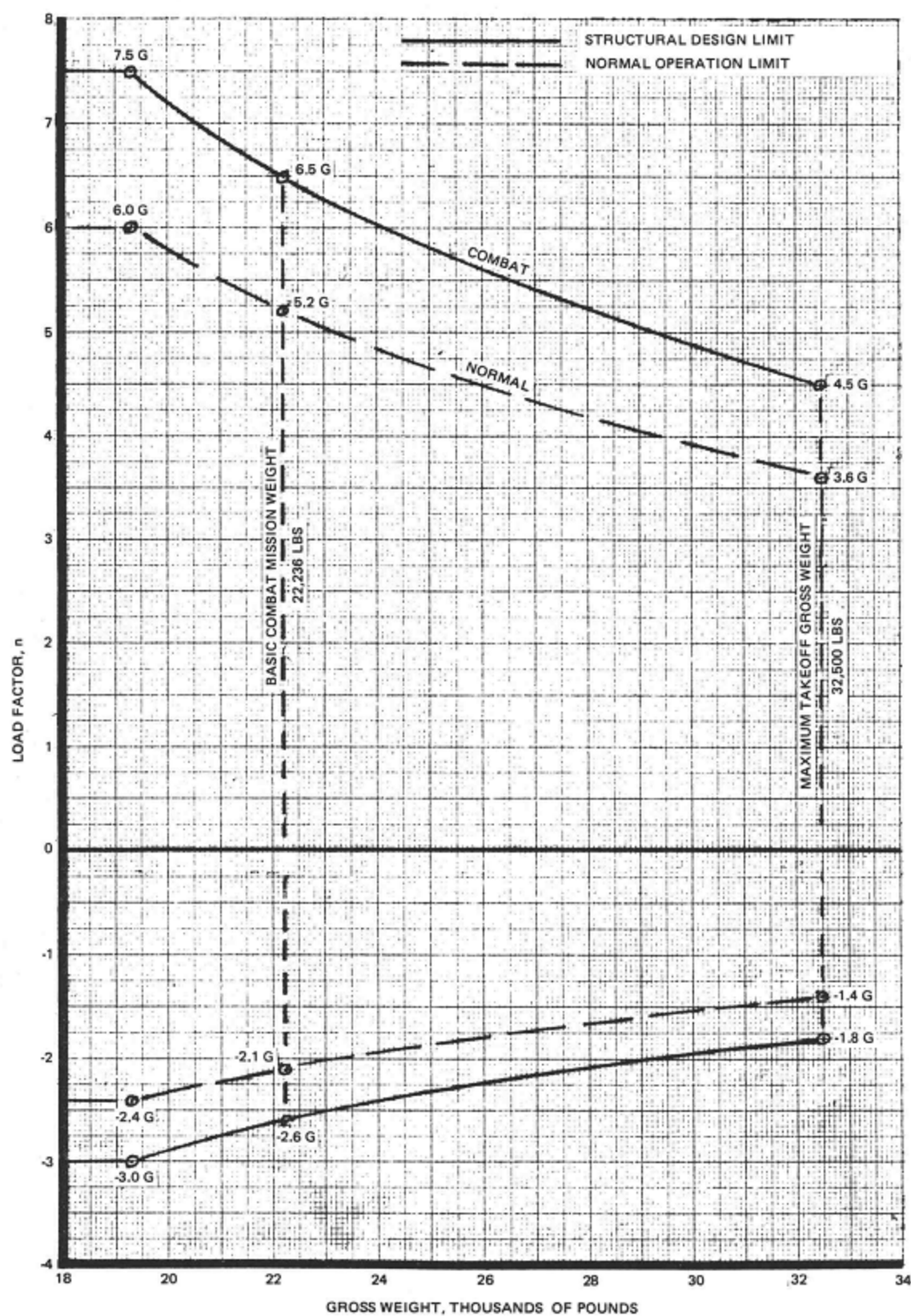


Figure 4-5. Design Load Factor Vs. Gross Weight

Fatigue analysis and testing is based on a scatter factor of 2 for fail-safe structure, such as the wing, and a scatter factor of 4 for safe-life structure, such as the tail surface pivot posts.

The maneuver loading spectrum of MIL-A-008866A, dated 31 March 1971, is modified to reflect a normal operations reduction of load factors above 6g. The positive load factor portion of the modified maneuver spectrum is shown on Figure 4-6, for the Lancer mission "mix" presented in Table 4-2. The modified spectrum is then used for analysis and fatigue testing. The design nW product for the Lancer is based on +6.5g, -2.6g at the Basic Design Combat Weight of 22,236 pounds. The maximum load factor is +7.5g, -3.0g at a lower weight, as shown in Figure 4-5.

The modification of the maneuver loading spectrum is accomplished by reducing the magnitude of all loadings above 6g (80 percent of 7.5). This modification still leaves 1000 loading applications of 6g or more (6 to 8.5g) which would cover inadvertent exceedances during normal operations. The actual measured data for F-104G operations at USAF Luke Air Force Base and Dutch Air Force Leeuwarden Air Base, converted to 4000 hours structural life, are also shown in Figure 4-6. The dashed line shows the unmodified spectrum, which would include one ultimate load per life.

The wing lower surface ultimate principal tension stress cutoff is calculated to be 48,000 psi, using the operational plan and the new criteria discussed above. The Lockheed method, described in Lockheed Structural Life-Assurance Memo No. 4, is used in the preliminary analysis for 4000 flight hours with the following criteria: A design quality index of 4, scatter factor of 2, the modified MIL-A-008866A maneuver spectrum discussed above, and fatigue data for 2024-T851 aluminum alloy plate material. Fuselage and tail surface ultimate tension stress cutoffs for aluminum alloy are similar to that for the wing. Cutoff stresses are also used for the titanium aftbody and empennage support structure.

4.1.2.4 Fail-Safe Criteria. The criteria for design of all fail-safe components require that, after failure of any single element, the remaining structure must be able to support static limit load.

4.1.2.5 Thermal Criteria. The thermal criteria for the Lancer structure are as follows:

- Surface temperatures based on 5 minutes of flight at Mach number 2.20
- Inlet duct design based on a maximum temperature of 312°F and 27 psi pressure



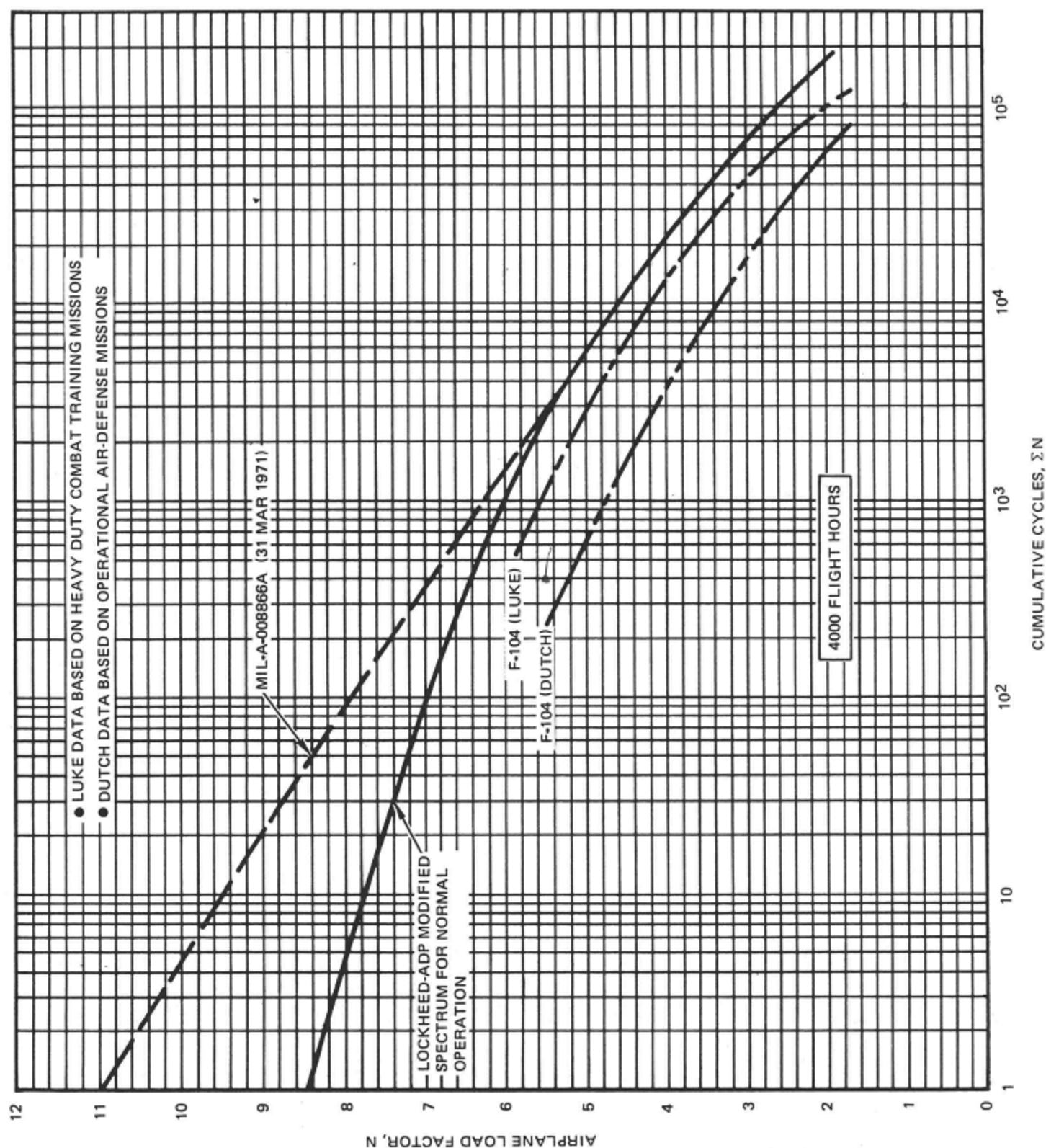


Figure 4-6. Modified Maneuver Spectrum - 4000 Hours

TABLE 4-2. MISSION MIX FOR FATIGUE ANALYSIS

MISSION	% TIME	APPROX. HRS/FLT	TOTAL HOURS
I FLIGHT OPERATIONS			
A LONG RANGE	10	2.0	400
B SHORT RANGE	50	1.0	2000
C OPER. TRAINING	20	1.0	800
II FLIGHT TRAINING			
A GENERAL	15	1.0	600
B LDG. PRACTICE	5	0.5	200
TOTALS	100		4000

- Exposure to temperature based on an accumulation of 100 hours of flight time at maximum Mach number (2.20) during the 4000 hours of airplane life
- Design load factors at various Mach numbers are shown on the Structural Design Speed Envelope, Figure 4-3.

4.1.2.6 Ground Loads Criteria. The Lancer uses the proven F-104S tricycle landing gear arrangement. As noted in Table 4-1, the landing design gross weight is 22,000 pounds with a center of gravity travel from 12 to 28 percent mean aerodynamic chord (m.a.c.). Design sinking speed at this weight is 10 feet per second (fps). The maximum takeoff gross weight and maximum landing weight is 32,500 pounds with a center of gravity travel from 12 to 18 percent m.a.c. Design sinking speed at this weight is 6 fps.

The ultimate factor of safety for all ground conditions including landing conditions is maintained as 1.5 times limit load. MIL-A-8862 criteria are considered to apply with the following exceptions:

- Drag reaction for the two point braked roll (Ref. para. 3.3.1.1) is based on a drag coefficient of 0.55, rather than 0.8.
- Value of "T" used in determining towing loads (Ref. para. 3.4.1) is based on 0.2W rather than the value from Figure 1 of MIL-A-8862.

4.1.3 Structural Materials

The types of materials and surface treatments used throughout the Lancer aircraft are summarized in Table 4-3.

TABLE 4-3. STRUCTURAL MATERIALS

MAJOR COMPONENT	ELEMENT	LOCATION	MATERIAL	TYPE/ALLOY	SURFACE * TREATMENT	
					INTERIOR	EXTERIOR
Fuselage	Skin	Sta.550.5 Fwd. Sta.550.5 Aft	Clad Alum. Sheet	2024-T81	1	2
	Longerons	Sta.550.5 Fwd.	Clad Alum. Sheet	2024-T81	1	
		Sta.550.5 Aft	Aluminum Extrusion Titanium Sheet	2024-T8510 6A1-4V Ann.	3 4,5,6	
	Rings-Frames	Sta.550.5 Fwd: Intermediate Rings Wing Attach. Frames	Clad Alum. Sheet Aluminum Forging	2024-T81 7049-T73	1 3	
		Sta.550.5 Aft: Intermediate Rings Tail Attach. Frame	Clad Alum. Sheet Titanium Forging	2024-T81 6A1-4V Ann.	1 4,5,6	
	Keelson	Sta.452-519.5	Aluminum Forging	7049-T73	3	
Wing	Inlet		Clad Alum. Sheet Aluminum Extrusion Aluminum Forging	2024-T81 2024-T62 7049-T73	1 3 3	2
	Surfaces	Upper-Lower	Bare Alum. Plate	2024-T851	3	3
	Beams	All	Aluminum Extrusion	2024-T8510	3	
	Ribs & Fittings	All	Aluminum Forging	7049-T73	3	
	L.E. Flaps	Skin	Bare Alum. Plate	2024-T851	3	3
	T.E. Flaps	Skin	Clad Alum. Sheet	2024-T81	1	2
	Aileron	Skin	Clad Alum. Sheet	2024-T81	1	2

* See Page 4-17

TABLE 4-3. STRUCTURAL MATERIALS (CONTINUED)

MAJOR COMPONENT	ELEMENT	LOCATION	MATERIAL	TYPE/ALLOY	SURFACE * TREATMENT	
					INTERIOR	EXTERIOR
Horizontal & Vertical Stabilizers	Surfaces		Clad Alum. Sheet	2024-T81	1	2
	Beams		Aluminum Extrusion	2024-T8510	3	
	Ribs		Clad Alum. Sheet	2024-T81	1	
	Fittings		Aluminum Forging	7049-T73	3	
	Pivot Posts		Titanium Bar	6Al-4V Ann.	4	
Landing Gear	Nose Gear	Main Strut	Alloy Steel	4340	7	
	Main Gear	Main Leg	Aluminum Forging	7049-T73	8	
Fasteners	Rivets	Aluminum Structure	Aluminum Alloy	2219-T81	9	
		Titanium Structure	Stainless Steel	A-286		
	Screws & Shear Bolts	Aluminum or Titanium Structure	Titanium & Stainless Steel Nuts	6Al-4V Aged A-286 (Nut)		
	Tension Bolts	Aluminum Structure	Stainless Steel Bolt & Nut	A-286		
		Titanium Structure	Titanium Bolt & Stainless Steel Nut	6Al-4V Aged 17-4PH(Nut)		
	Spotwelds	Aluminum Structure Titanium Structure	No Spotwelds in Primary Structure Maximum of Two Sheets			

* See Page 4-17

TABLE 4-3. STRUCTURAL MATERIALS (CONTINUED)

NOTES: SURFACE TREATMENT

1. a) Chemical Conversion Treatment per MIL-C-5541
b) Zinc Chromate Primer, MIL-P-8585
2. a) Chemical Conversion Treatment per MIL-C-5541
b) Wash Primer, MIL-C-8514
c) Intermediate Lacquer Primer, MIL-P-7962
d) First Top Coat, Acrylic Lacquer, MIL-L-19537
e) Camouflage Top Coat, Acrylic Lacquer, MIL-L-19538
3. a) Sulphuric Acid Anodize and Dichromate Seal per MIL-A-8625, Type II
b) 100% Shot Peen before Anodizing and Sealing
4. NO FINISH
5. Titanium-to-Aluminum Joints:
a) Zinc Chromate Primer per MIL-P-8585 on Each Surface
b) Fay Surface Seal with 3M-XA 5147 B/A Sealant
6. Titanium-to-Steel Joints:
a) Phosphate Treat Steel
b) Zinc Chromate Primer per MIL-P-8585 on Each Surface
c) Fay Surface Seal with 3M-XA 5147 B/A Sealant
7. a) Coat with MIL-H-27601 or Rust Lick 606
b) Bake for 1 hour at 450°F.
c) Finish with Flexible Epoxy-Polyurethane
8. a) 100% Shot Peen
b) Sulphuric Acid Anodize
c) Finish with Flexible Epoxy-Polyurethane
9. a) All Fasteners Are Installed Wet with 3M-XA 5147 B/A Sealant

Aluminum alloy materials in the Lancer are chosen not only for high strength consistent with superior resistance to stress corrosion, but also with regard to temperature environment. These considerations dictate the use of 2024-T81 sheet, 2024-T851 plate, 2024-T8510 extrusion, and 7049-T73 forgings as the basic structural material. The Lancer structure is more than 90% aluminum alloy.

Titanium alloy is employed primarily in the aft fuselage where high temperature, fire resistance, or structural efficiency precludes the use of aluminum. Annealed 6Al-4V alloy is chosen on the basis of strength, corrosion resistance, proven service history, and availability in production quantity. Titanium alloy makes up approximately 6 percent of structure weight of the airplane.

4.1.4 Aeroelasticity

4.1.4.1 Flutter Analysis. The Lancer flutter analysis used the station method in which the stiffness, inertia, and aerodynamic forces are concentrated at points located on the lifting surface.

Flutter analyses for all lifting surfaces were conducted at Mach numbers 0.95, 1.25 and 2.00. The most critical flutter cases for the lifting surfaces are summarized in Figures 4-7, 4-8, 4-9 and 4-10, indicating safe flutter margins. The flutter results for the supersonic Mach numbers show much larger flutter margins due to the decreased lift curve slope of the wing and rearward aerodynamic center shift.

The flutter analyses results show no difference in flutter speed for the configurations with or without wing internal fuel. Therefore, the results for the no-fuel cases are shown in Figures 4-7 through 4-10. The wing bending and torsional stiffness distribution used in the aeroelastic analyses are shown in Figure 4-11.

The all-movable horizontal and vertical stabilizers are analyzed taking into account mounting posts, control systems, actuators, and backup structure. Two configurations were studied for each stabilizer: first, the normal control system with two actuators effective, and second, with only one actuator effective.

4.1.4.2 Aileron Effectiveness. Flexible aileron effectiveness versus Mach number at various altitudes is shown in Figure 4-12. The analysis is based on wing stiffness parameters given in Figure 4-11. The effect of aeroelasticity on aileron effectiveness was included in all stability and control analyses of the Lancer.

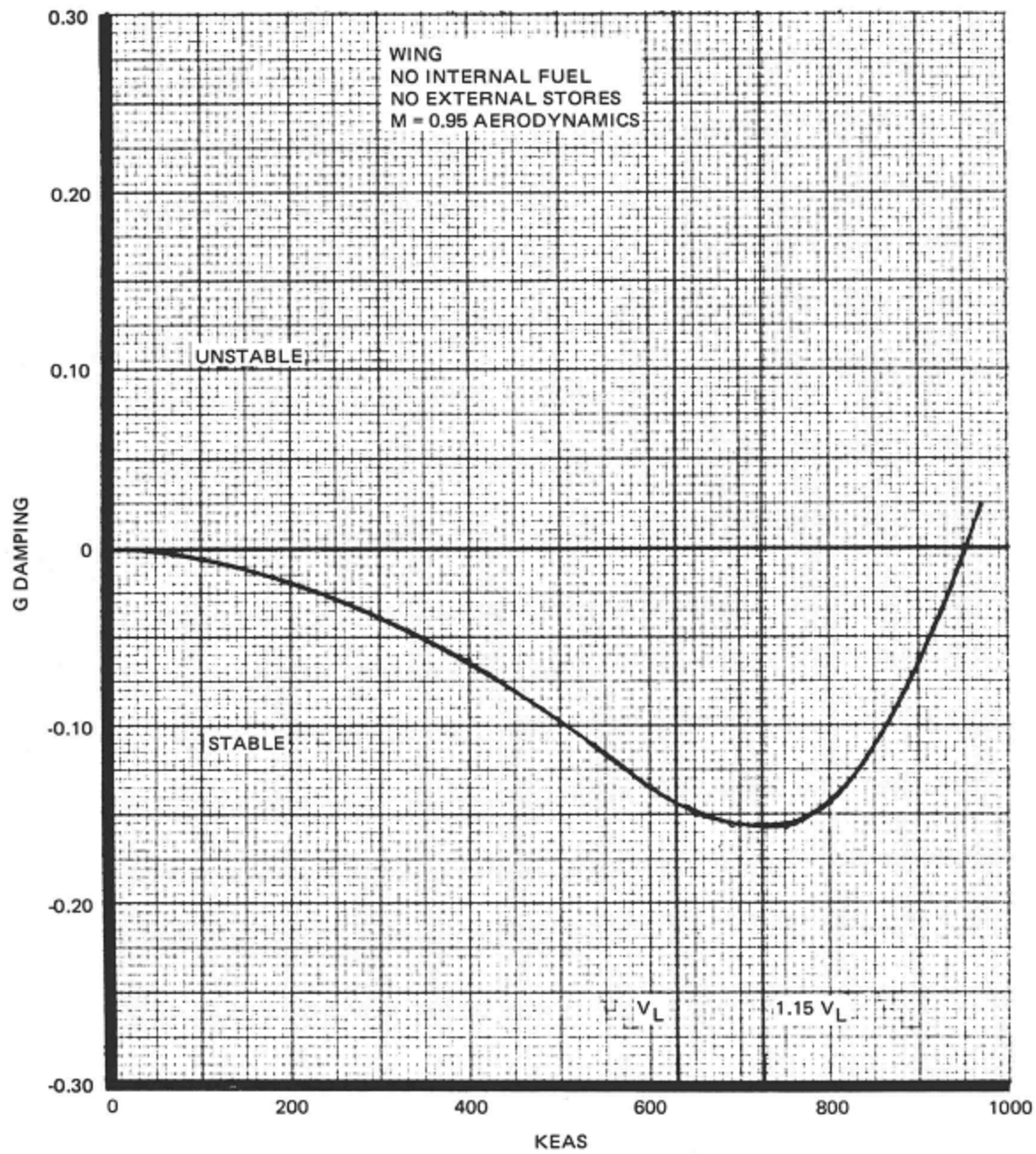


Figure 4-7. Wing Transonic Flutter

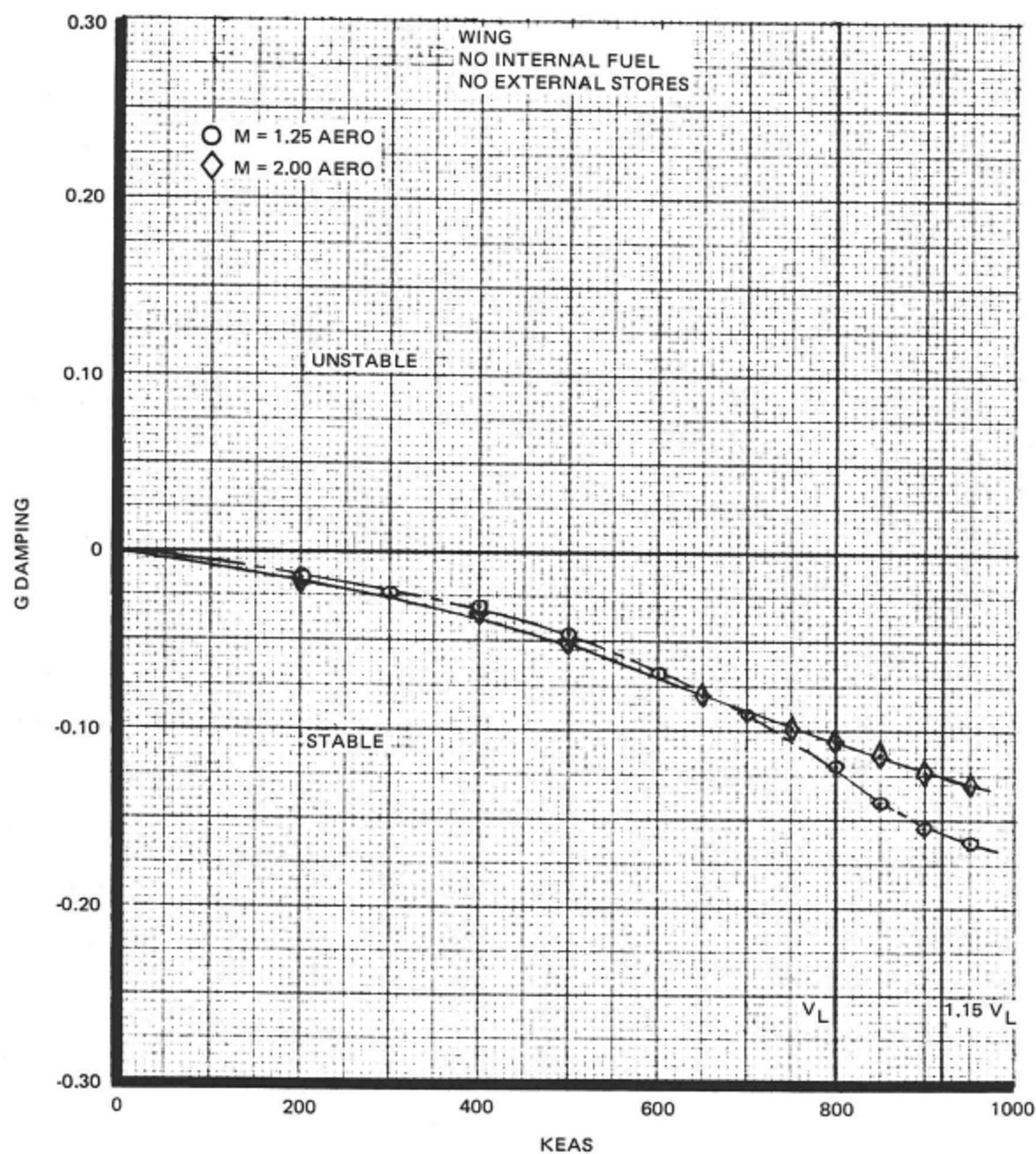


Figure 4-8. Wing Supersonic Flutter

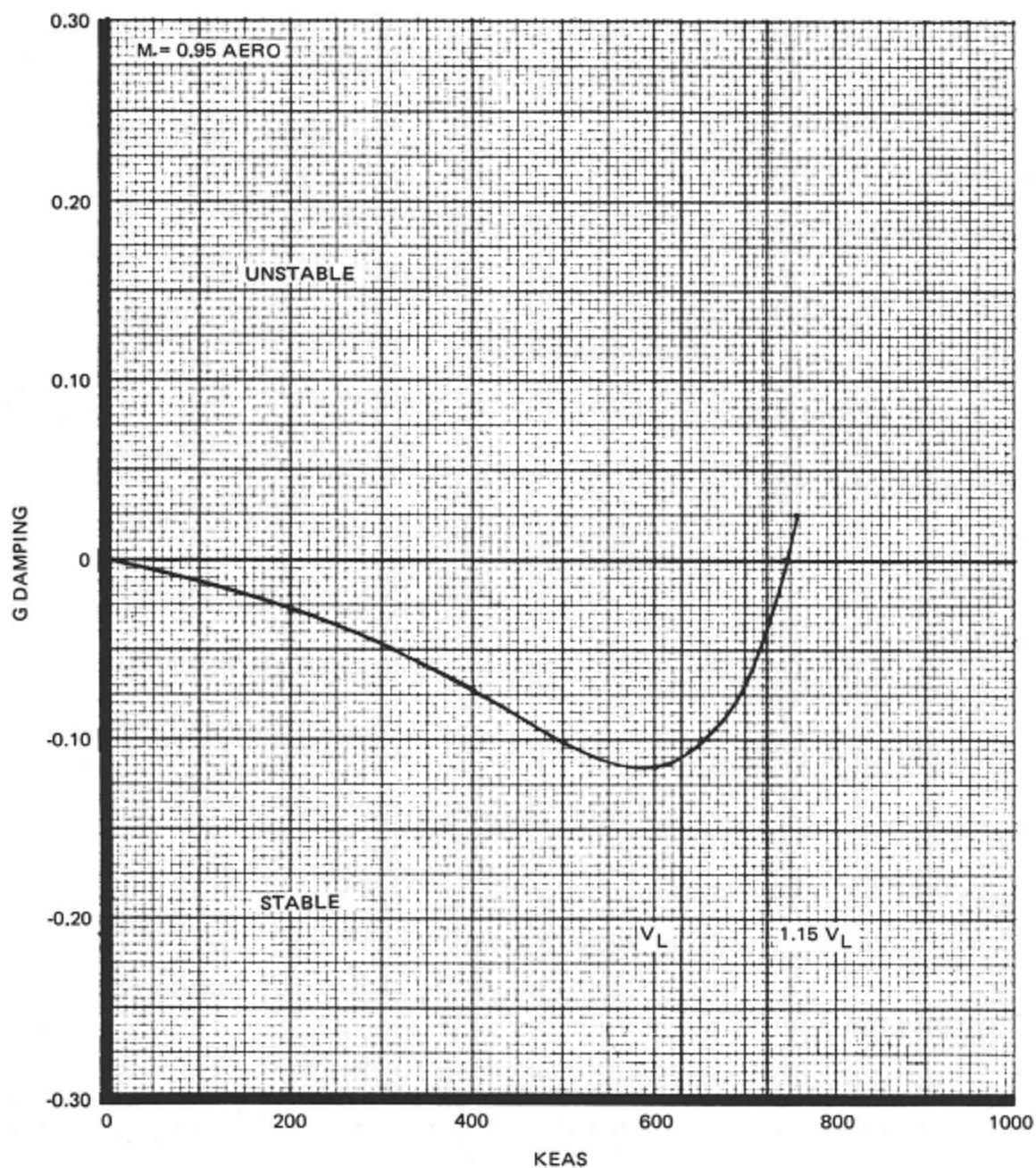


Figure 4-9. Horizontal Stabilizer Transonic Flutter

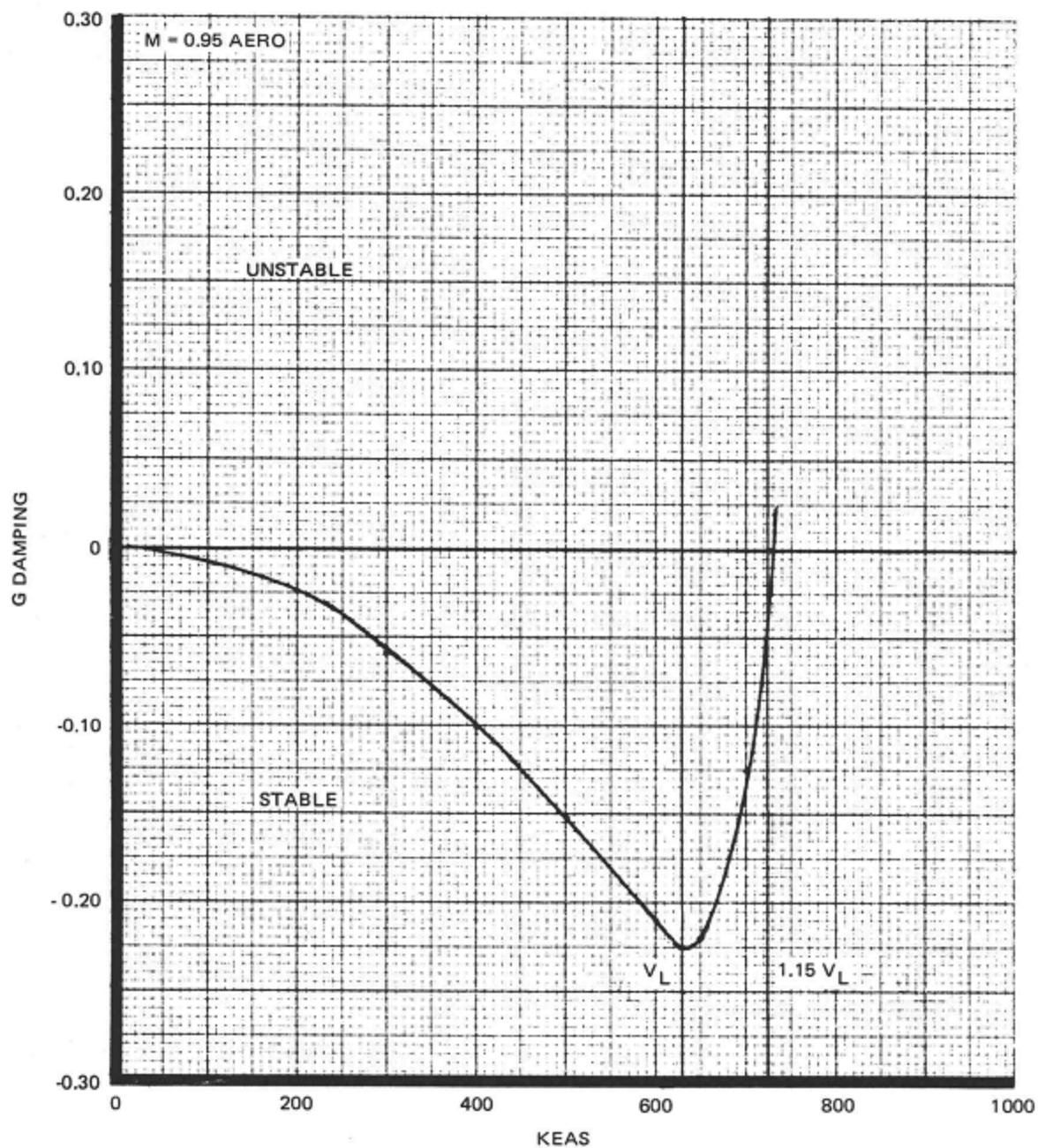


Figure 4-10. All Movable Rudder Transonic Flutter

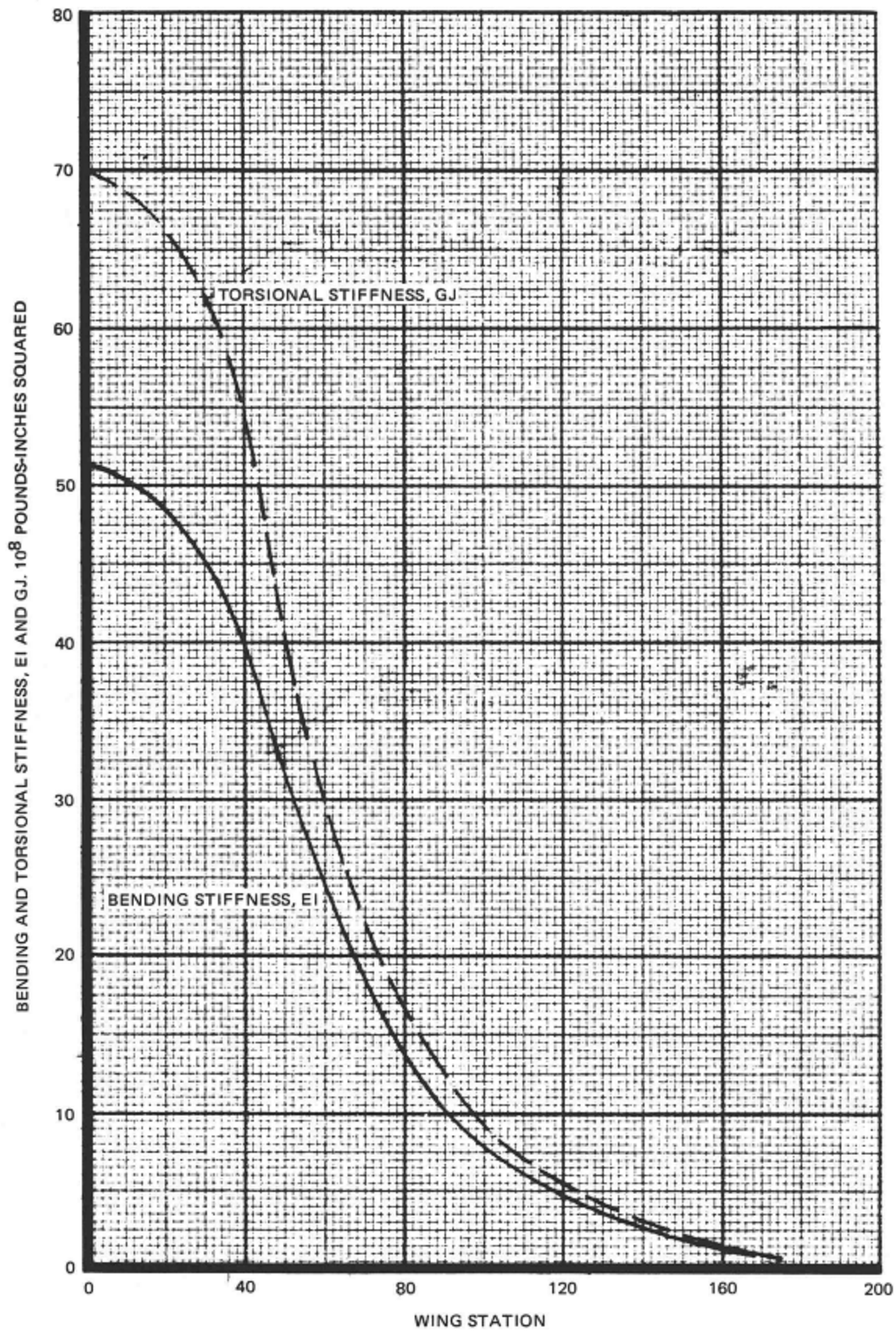


Figure 4-11. Wing Box Stiffness

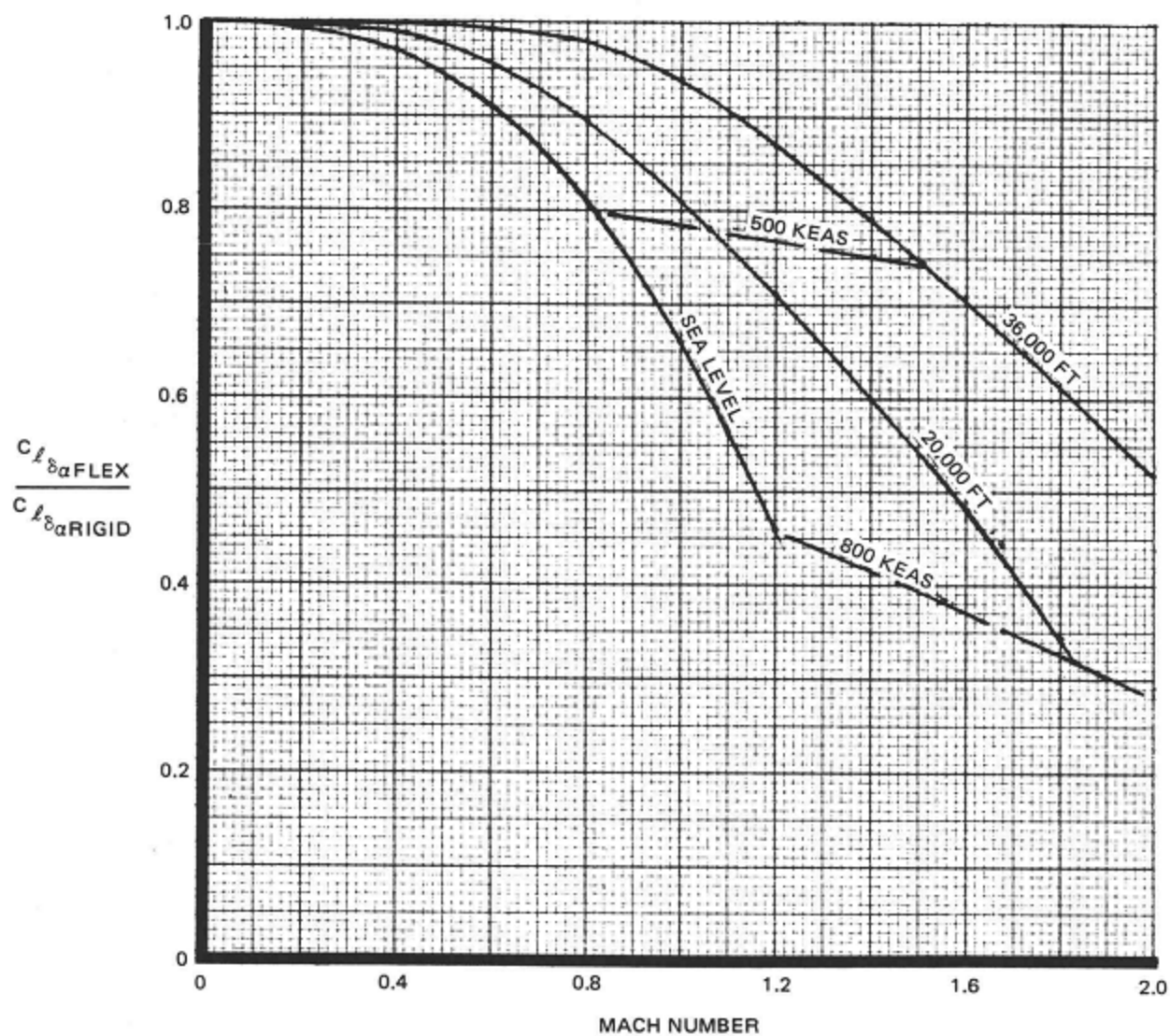


Figure 4-12. Aileron Effectiveness

4.1.5 Weight and Balance

The group weight breakdown for the clean zero fuel weight of the Lancer is shown in Table 4-4.

The takeoff gross weights for various typical missions are shown in Table 4-5.

Typical center-of-gravity diagrams for the Basic Air Superiority Mission are shown in Figure 4-13, for the Close Air Support Mission in Figure 4-14, for the Sea Level Bombing Mission in Figure 4-15, for the Ferry Mission in Figure 4-16, and for a mission with the Sparrow missiles in Figure 4-17. These diagrams also show the center-of-gravity envelopes defined by the structural and aerodynamic limits of center-of-gravity travel and weight. The aerodynamic limits have been established by wind tunnel tests of the Lancer configuration.

TABLE 4-4. WEIGHT BREAKDOWN

GROUP	WEIGHT, LB.
Wing	2480
Tail	940
Fuselage	3100
Landing Gear	970
Propulsion and Inlets	5890
Surface Controls	650
Instruments	90
Hydraulics	300
Electrical	350
Avionics	700
Armament Provisions	370
Furnishings	178
Air Conditioning	325
Deceleration Chute	48
Contingencies	100
Weight Empty	16491
Unusable Fuel	90
Oxygen	13
Pilot	210
Engine Oil	30
Gun (Linkless Feed) 20 MM	131
Ammo 500 Rounds: Retained Cases	135
: Expendable	150
Clean Zero Fuel Weight	17250

TABLE 4-5. MISSION TAKE-OFF WEIGHTS

ITEM	AIR SUPERIORITY COUNTER AIR DASH INTERCEPT POINT INTERCEPT	SEA LEVEL BOMBING	CLOSE AIR SUPPORT	FERRY	SUBSONIC POINT INTERCEPT
Clean Zero Fuel Weight	17250	17250	17250	17250	17250
Internal Fuel	9075	9075	9075	9075	9075
2 Wing Tip Launchers	110		110		
2 AIM-9 Sidewinders	310		310		
5 MK-83 Bombs		5000			
2 W.S. 152 Bomb Pylons		242	242		
2 W.S. 112 Bomb Pylons		324	324		
C _L Fuselage Bomb Rack		127	127		
5 M-117 Bombs			4000		
2 Wing Tip Fuel Tanks				460	
Wing Tip Tank Fuel				2210	
2 Pylon Fuel Tanks				356	
2 Pylons for Fuel				275	
Pylon Tank Fuel				2535	
2 Wing Tip Sparrows					900
2 Wing Tip Launchers					101
2 Outboard Pylon Sparrows					900
2 Outboard Pylons & Launchers					433
2 Inboard Pylon Sparrows					900
2 Inboard Pylons & Launchers					414
T.O.G.W. (LBS.)	26745	32018	31438	32161	29973

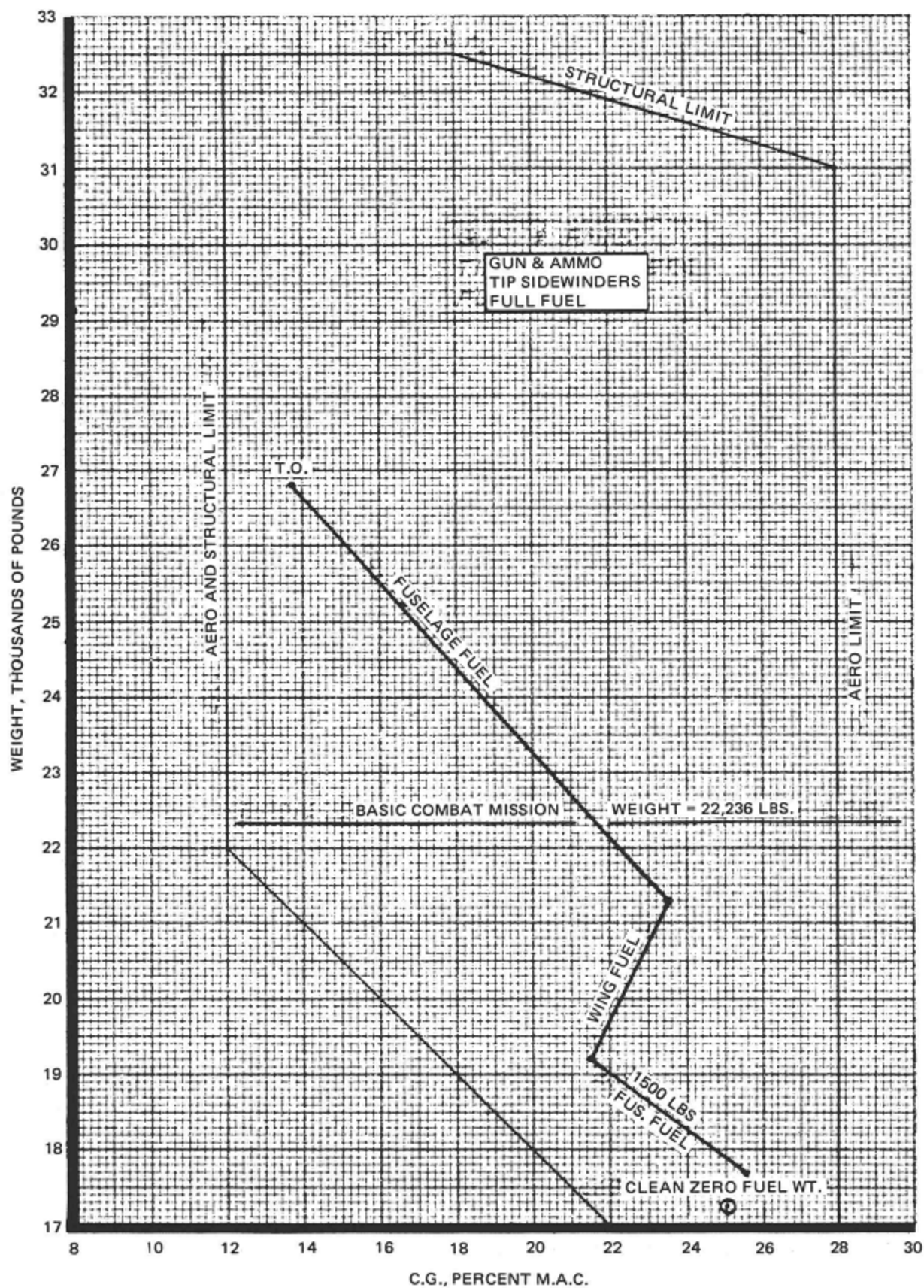


Figure 4-13. CG Diagram - Air Superiority Missions

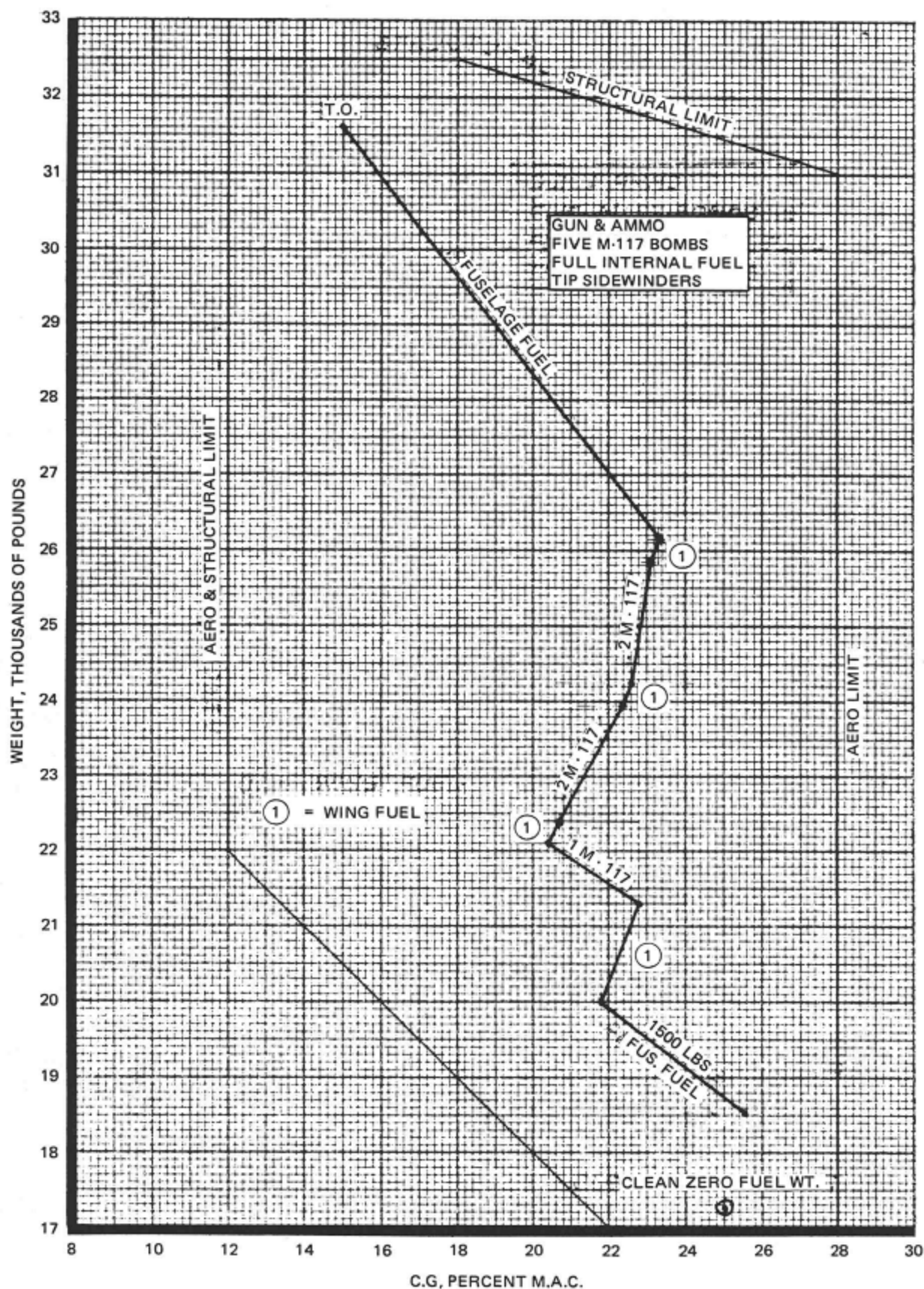


Figure 4-14. CG Diagram - Close Air Support Mission

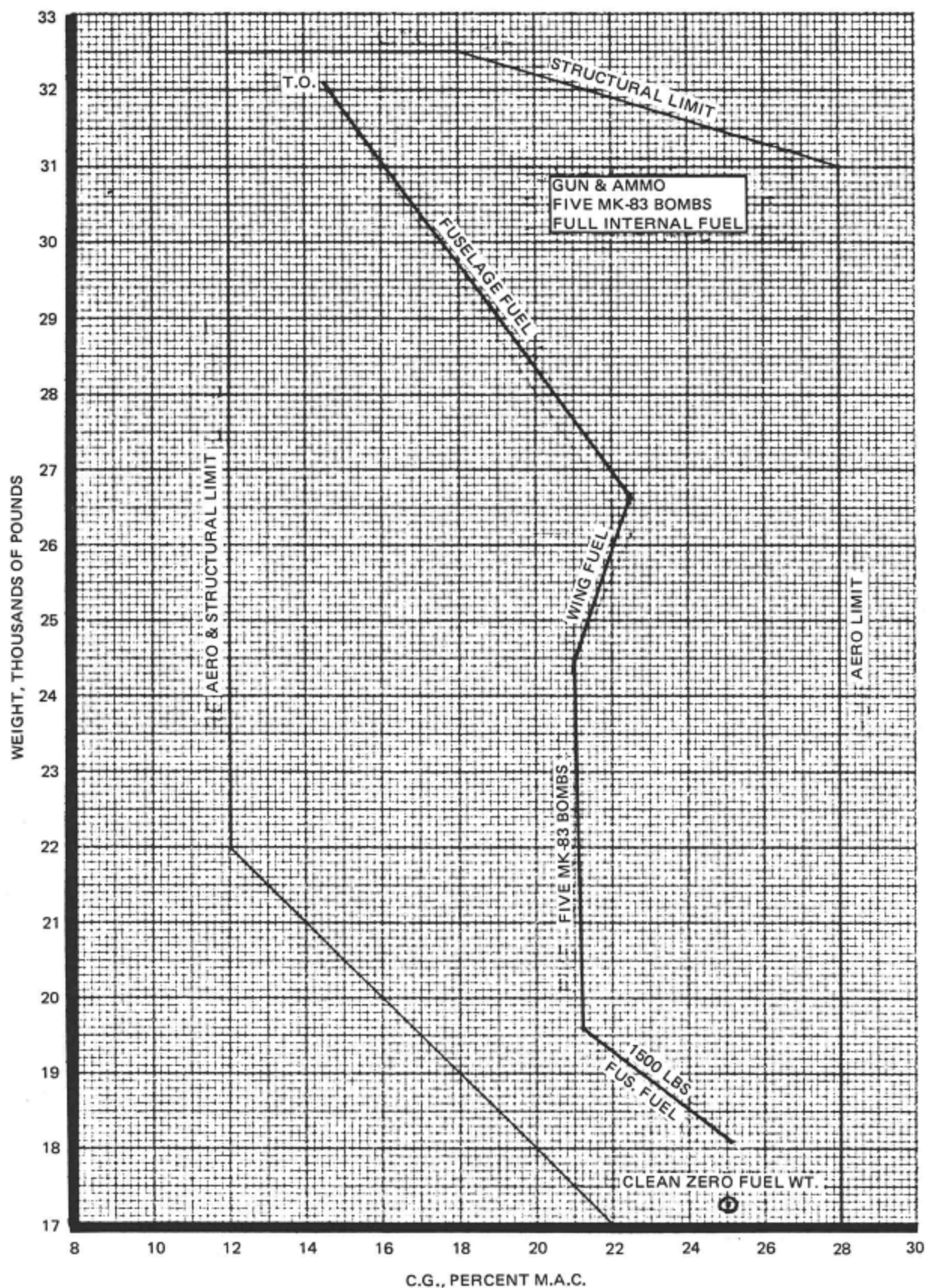


Figure 4-15. CG Diagram - Sea Level Bombing Mission

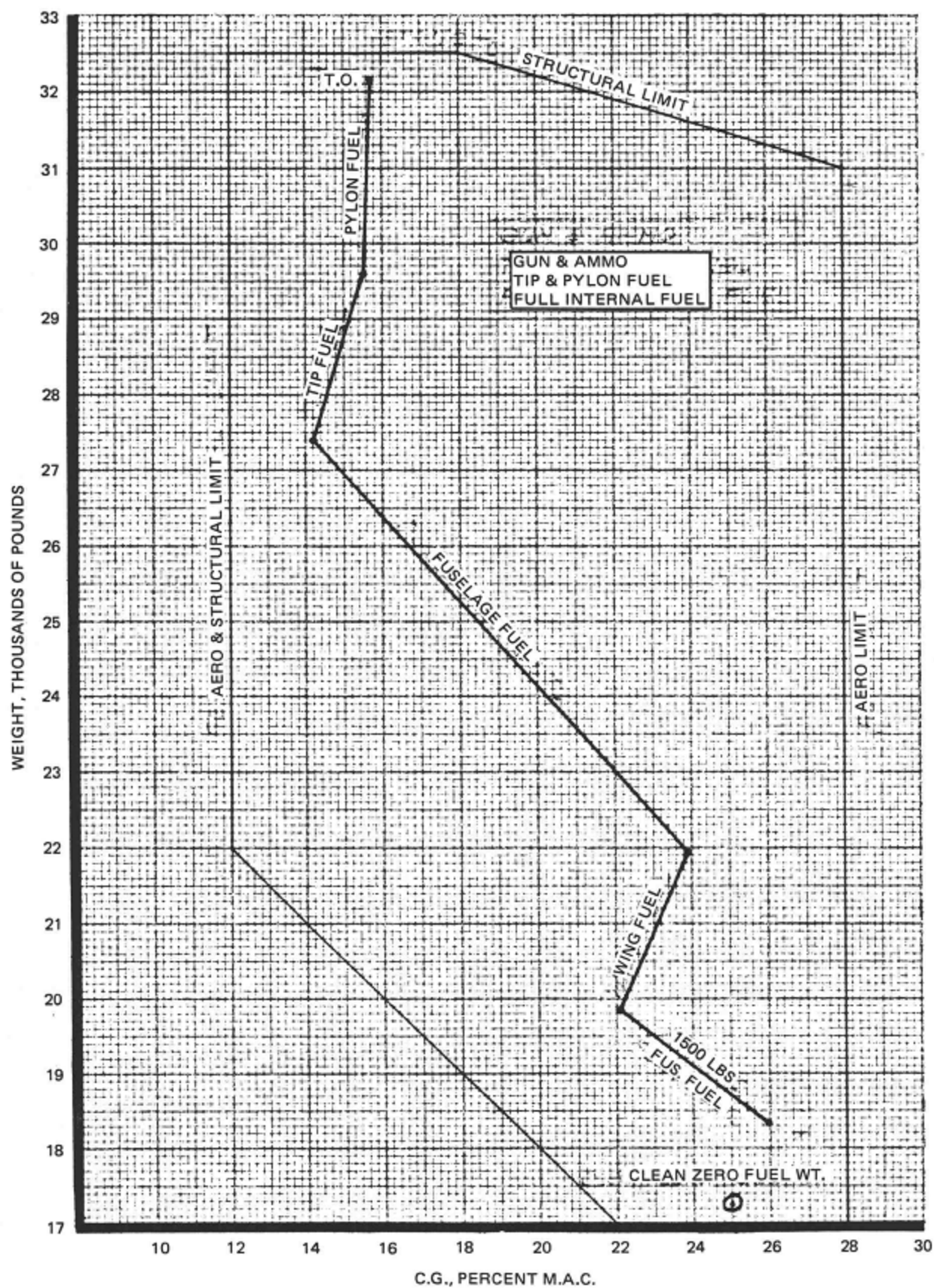


Figure 4-16. CG Diagram - Ferry Mission

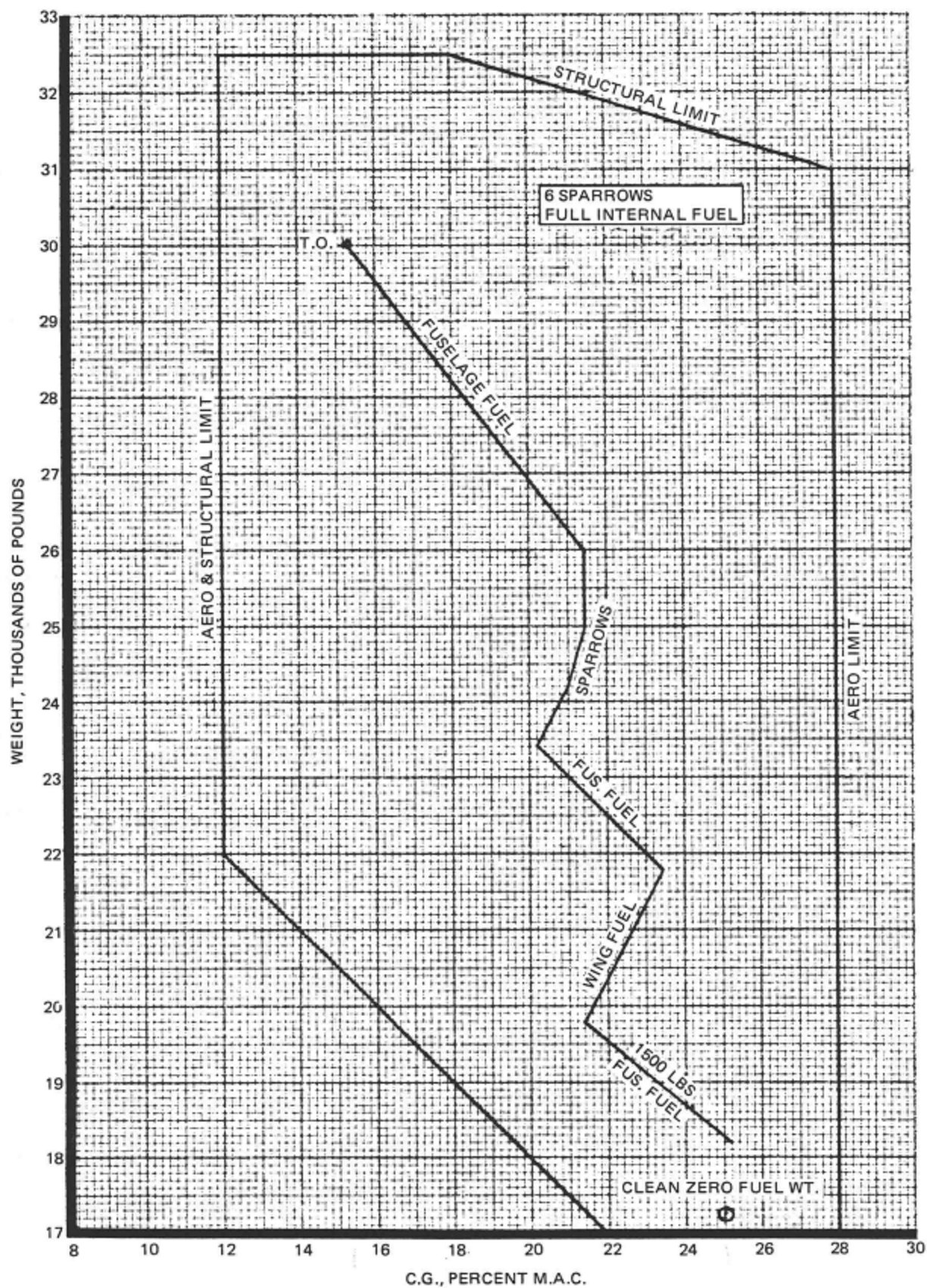


Figure 4-17. CG Diagram - Subsonic Point Intercept Mission

4.2 POWERPLANT

The powerplant for the Lancer is the Pratt & Whitney JTF10A-40, a single engine aircraft version of the TF30-P-100 engine described by Pratt & Whitney Specification A-6124 and presently operational in the F-111F aircraft. The JTF 10A-40 is an augmented two spool axial flow gas turbine engine. A full annular fan air duct surrounds the basic gas generator and supplies fan air to the mixed-flow augmentor. The engine is supplied with a variable area convergent exhaust nozzle. Blow-in doors and the ejector nozzle are built into the aft end of the fuselage, allowing efficient use of inlet matching bypass airflow and facilitating engine removal. During low speed operation when bypass airflow is not available, suck-in doors in the engine compartment provide engine cooling.

An accessory gearbox supplied on the engine provides pads for two hydraulic pumps, an integrated constant speed drive/generator, and a tachometer generator. Access for servicing the engine accessories main fuel feed line and throttle quadrant is through a hinged door at the bottom of the airplane.

A self-contained jet fuel starter is connected to the engine starter pad. This starter provides ground starts and air start assistance up to 15,000 feet. Once initiated, starter operation is automatic.

Engine anti-icing is provided on the inlet guide vanes using twelfth stage bleed air. A fire detection system is provided and consists of two sets of continuous loops located at appropriate positions within the engine compartment.

A manual fuel control provides backup capability for nonaugmented engine operation in the event the primary hydromechanical fuel control malfunctions.

The TF30-P-100 engine provides 40 percent greater thrust at maximum power than the J-79-19 with a lower engine weight and lower cruise SFC.

4.3 FUEL SYSTEM

The fuel tanks of the Lancer are listed and their capacities shown in the following tabulation.

<u>Internal:</u>	<u>Gallons</u>	<u>Pounds</u>
Fuselage sump tank	747	4,856
Fuselage aft tank	326	2,119
Wing Tank	<u>323</u>	<u>2,100</u>
Total Internal	1,396	9,075
 External:		
Wing-tip tanks	340	2,210
Wing pylon tanks	<u>390</u>	<u>2,535</u>
Total External	730	4,745
Grand Total	2,126	13,820

A schematic drawing of the fuel system is presented in Figure 4-18. Both fuselage fuel tanks are integral tanks. The main fuselage tank is used as the sump from which two electrically driven booster pumps supply fuel to the engine driven pump. Fuel from the fuselage aft tank, surrounding the engine inlet air ducts, transfers to the sump tank by gravity through valved interconnectors. Fuel in the integral wing tank is transferred automatically to the sump tank by an electrically driven pump when the fuel level in the sump tank is lowered to a predetermined quantity. This arrangement provides optimum cg control during flight. Fuel from the wing tip and wing pylon tanks (provided for ferry and other external fuel missions) is transferred to the sump tank by regulated engine bleed air pressure through the float-actuated refueling shutoff valve. The external tanks are installed or removed easily and can be jettisoned in flight. Fuel transfers from the pylon tanks first when both tip and pylon tanks are installed, and the tip tanks transfer automatically when pylon tanks are empty or jettisoned.

Single-point pressure fueling permits filling all tanks simultaneously. Fuselage tanks and external tanks can also be filled through conventional gravity fillers. The Lancer is designed to use the F-104's tip and pylon tanks.

A cooling loop through an air-to-fuel heat exchanger is pressurized by a separate circulation pump, identical to the wing transfer pump, and is fed from the boost pump manifold. Hot fuel is routed to the engine feed manifold. A hydraulic oil-to-fuel heat exchanger loop is tapped off the engine feed line and warm fuel is returned to the fuel tank.

The sump and aft tanks contain quantity measuring probes connected to a cockpit indicating gage. Wing and external tank fuel quantities are not included on the gage, but lights on the instrument panel give indications of WING TANK FULL, WING TANK EMPTY, TIP TANK EMPTY, FUEL LEVEL LOW, and FUEL BOOST PUMP FAIL when energized by float or pressure switches within the fuel tanks.

4.4 FLIGHT CONTROLS AND HYDRAULIC SYSTEM

4.4.1 Primary Flight Controls

The primary flight controls use single cable systems and pushrods for transmittal of pilot inputs, as shown in Figure 4-19. Each system is routed from the cockpit to the servos by the most direct path to reduce friction to approximately 2 pounds. This keeps breakout forces to a minimum and contributes to the aircraft's excellent handling qualities. Artificial feel and centering forces, provided by fail-safe cartridge-type springs located near the servo input mechanism, are optimized for light control response while maintaining a positive feel.

Irreversible servos with dual hydraulic systems are located near the control surfaces and provide redundancy for safety of flight (see paragraph 4.4.3). Pitch and roll trim are accomplished by series electrical actuators while yaw trim is provided electronically through the yaw SAS system. The trim systems are located in the feedback loop so that a backup mode for safety of flight is accomplished in the event of a control cable or pushrod failure. Surface trim position is not reflected in the stick or pedals.

The Lancer uses all-movable horizontal and vertical stabilizers. The horizontal stabilizer and the large upper portion of the vertical stabilizer move as complete control surfaces to provide pitch and yaw control. U.S. military specifications and standards are used as the design objective for the entire flight control system insofar as consistent with the primary objectives of lightweight and low cost.

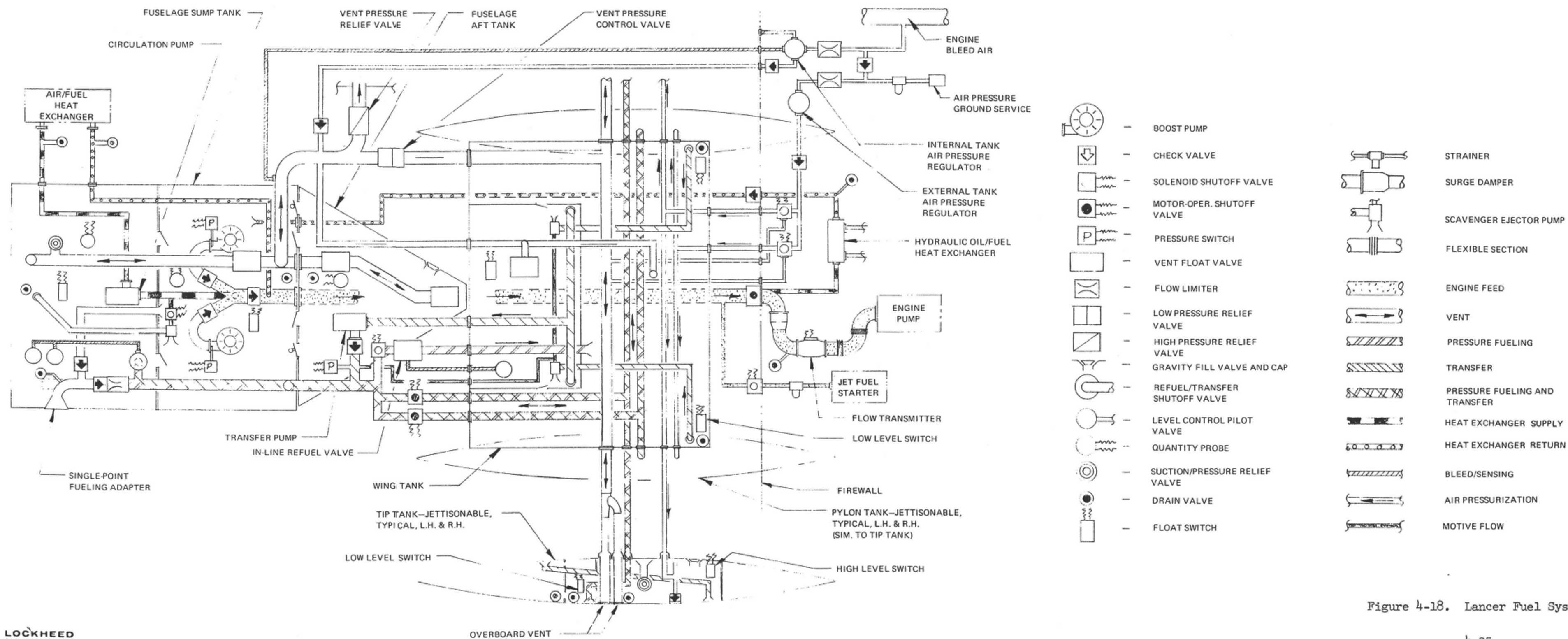


Figure 4-18. Lancer Fuel System

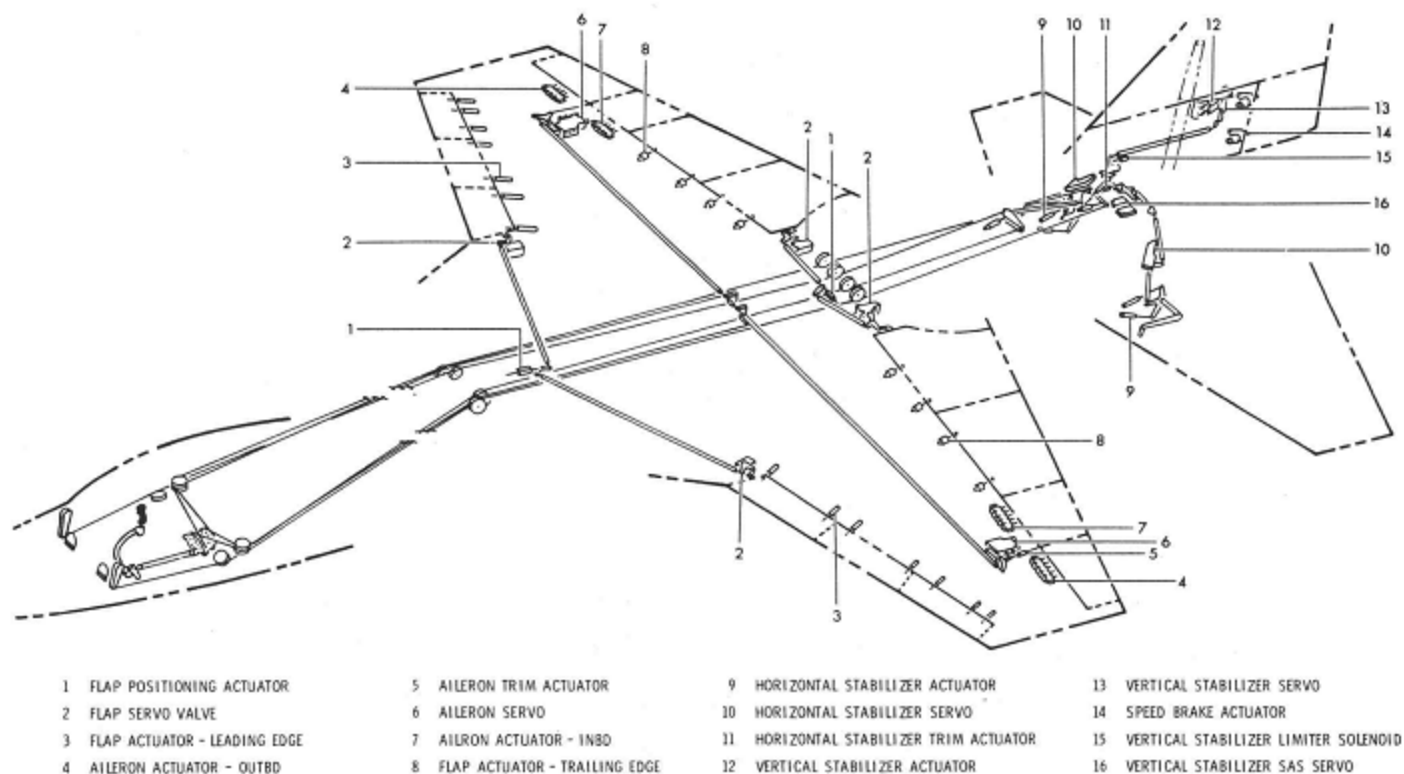


Figure 4-19. Flight Controls

The movable vertical stabilizer develops large yawing forces for much improved combat agility. The normal $\pm 6^\circ$ surface travel of the vertical stabilizer is increased to $\pm 20^\circ$ when the landing gear is extended to enhance cross wind landing capability. The full travel of the horizontal stabilizer is 5° leading edge up to 25° leading edge down.

The leading and trailing edge flaps are hydraulically powered, but are controlled electrically. Symmetry of these surfaces and those of the horizontal stabilizer is assured by interconnecting the servo-mechanical linkages across the aircraft. Hydraulically-operated speed brakes may be adjusted to any position for precise speed control at combat speeds (see paragraph 4.5.3).

4.4.2 Automatic Flight Control System (AFCS)

The Automatic Flight Control System consists of a three-axis Stability Augmentation System (SAS), automatic Mach trim and a fully automatic maneuvering flap system. Command Augmentation Steering (CAS) is included as a basic mode of the pitch and roll SAS. Acceleration and rate inputs are used by CAS to provide more uniform stick force and handling characteristics throughout the airspeed, altitude, and Mach number limits of the flight envelope.

Automatic Mach trim enhances speed stability and the automatic maneuvering flap system positions leading and trailing edge flap angles as a function of Mach number, dynamic pressure and load factor. All required dynamic pressure and Mach number, altitude and KEAS schedules for the AFCS are supplied by the central Air Data Computer (ADC). Load factor for the automatic maneuvering flap system is sensed by vertical accelerometers.

The pitch and roll AFCS are mechanized for fail operational performance. The pitch CAS provides static stability augmentation to result in uniform and improved handling qualities throughout the flight envelope. To preclude unintentionally overstressing the airplane, the pitch CAS provides a threefold increase in stick force-per-g when the limit load factor is approached. The roll CAS produces a nearly constant roll rate per pound of stick force. Rudder pedal position is crossfed to the roll SAS to provide normal aileron/rudder "feel" characteristics at all Mach numbers and altitudes. This crossfeed is disconnected when the landing gear is lowered, to increase cross wind landing capability. Simplified block diagrams of the pitch and roll AFCS are shown in Figures 4-20 and 4-21.

The single channel yaw SAS provides improved turn coordination and damping of the Dutch roll mode. Lateral acceleration is disconnected upon rudder pedal input to allow the pilot to intentionally miscoordinate the airplane when maneuvering in combat or when making a cross wind landing. Manual yaw trim is accomplished through the yaw SAS, which is depicted in Figure 4-22.

4.4.3 Hydraulic System

As shown in Figure 4-23, there are two independent 3000 psi hydraulic systems, each of which has its own engine-driven hydraulic pump. The No. 1 system powers the primary flight controls exclusively. The No. 2 system powers all the hydraulic services on the aircraft, including the primary flight controls. Hydraulic pressure for the primary flight controls is protected by a priority valve. Hydraulic flow diagrams for the two systems are shown in Figures 4-24 and 4-25. The No. 2 system has a larger pump since the flow requirements are greater for this system.

The flight controls may also be powered by the ram air turbine, which can be extended into the airstream within 5 seconds to provide a 6-gpm flow to the No. 1 system. The turbine is mechanically extended into the airstream by pilot control and/or by loss of engine rpm. This emergency power source provides the required surface contro

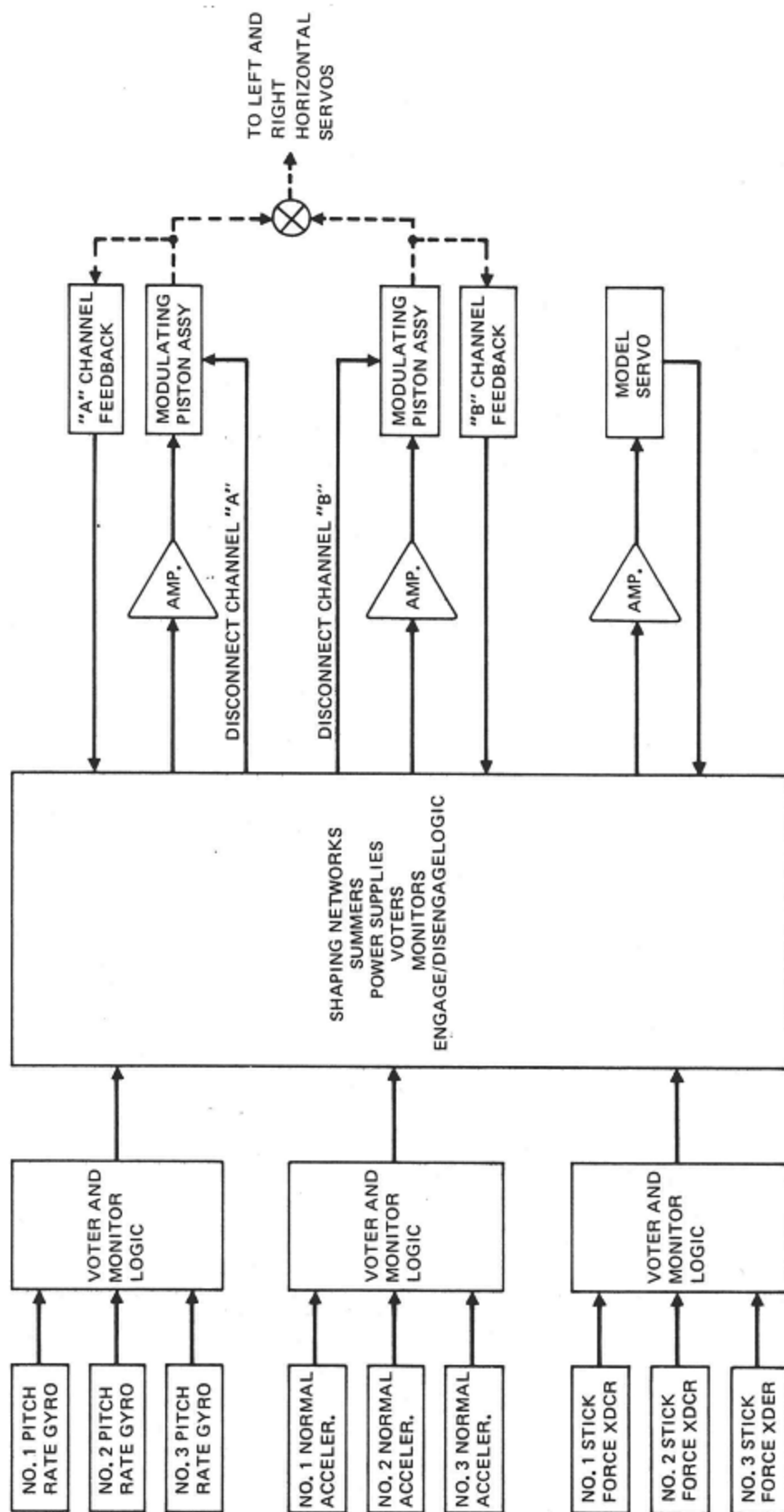


Figure 4-20. SAS Schematic - Pitch

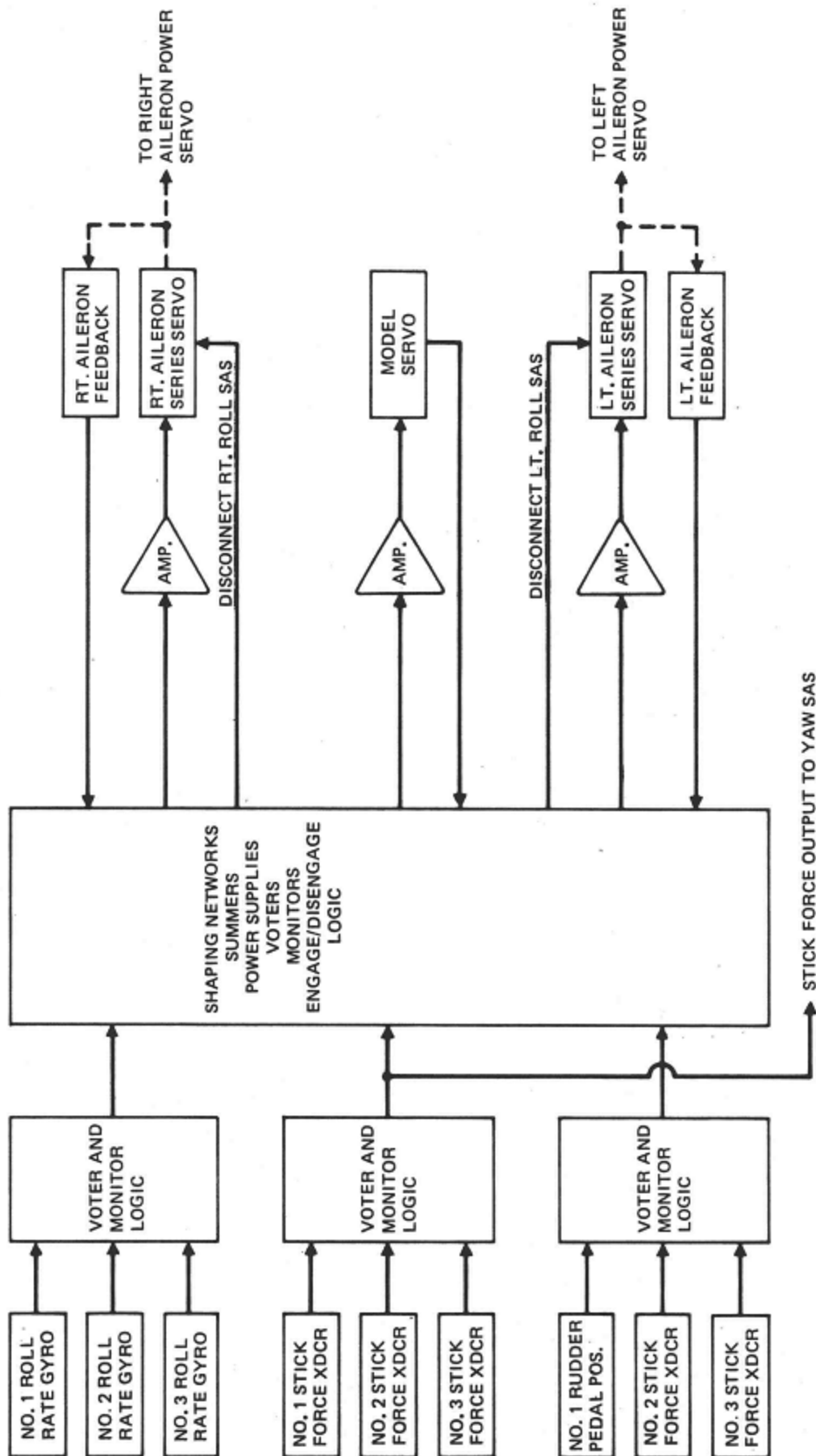


Figure 4-21. SAS Schematic - Roll

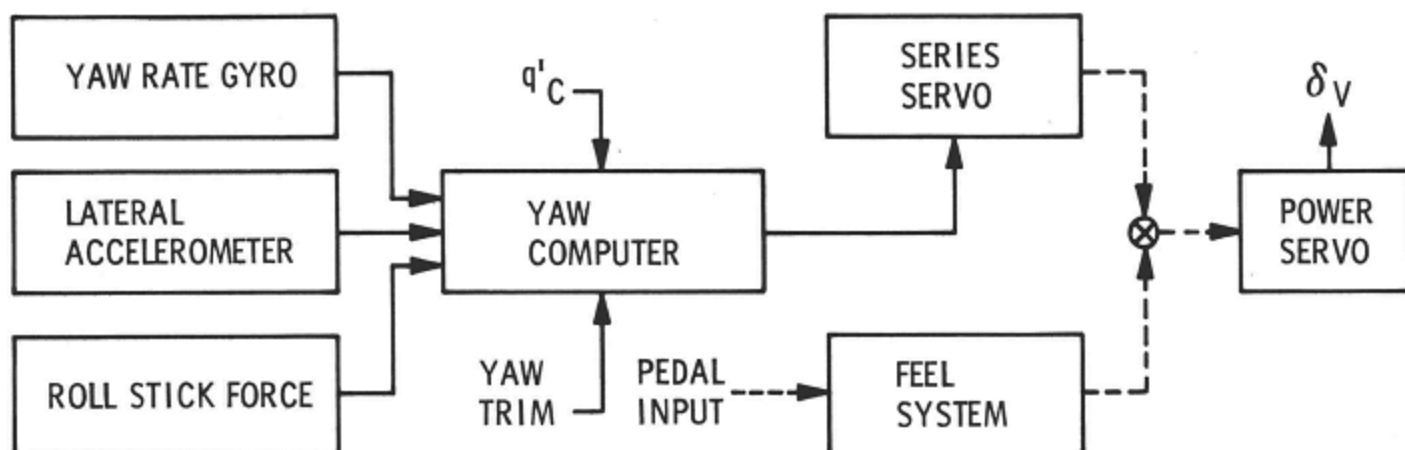


Figure 4-22. Yaw SAS Diagram

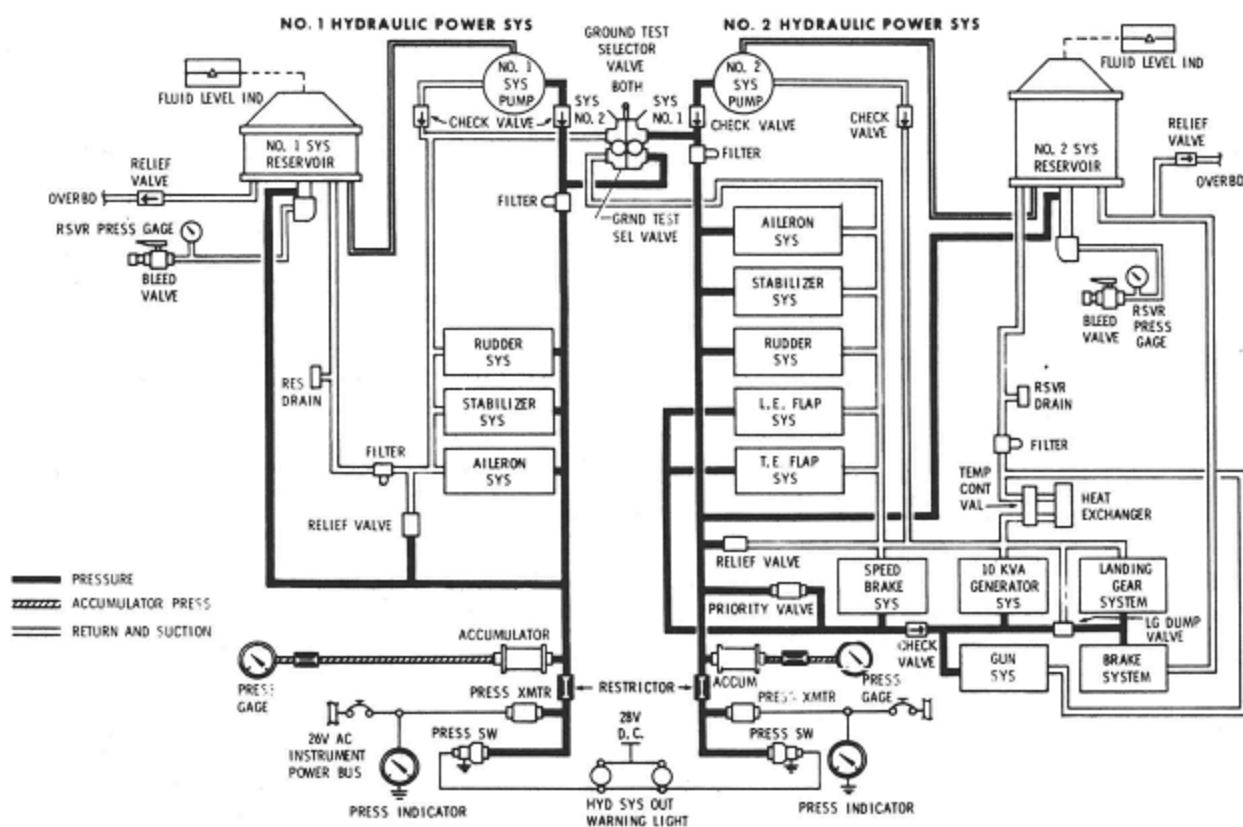


Figure 4-23. Hydraulic Schematic

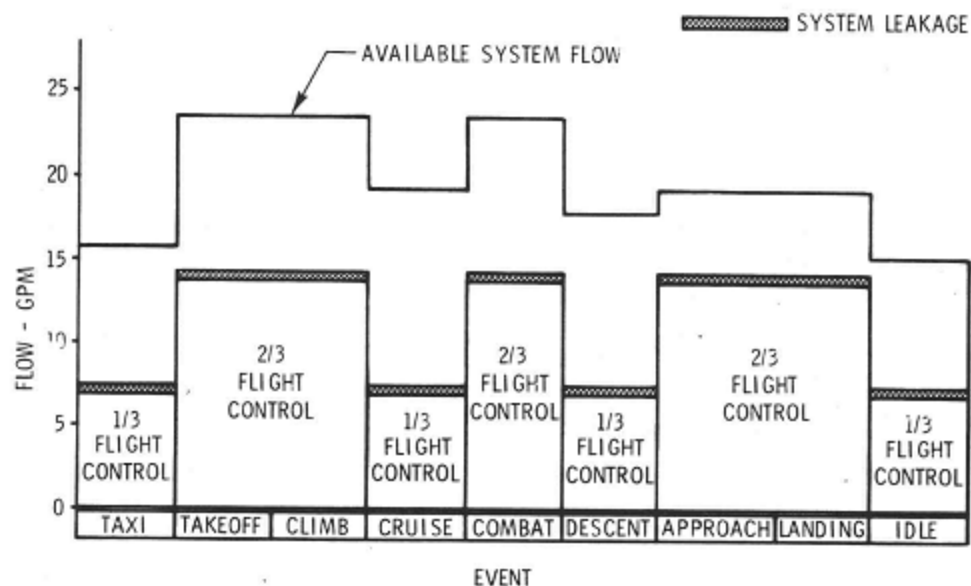


Figure 4-24. Hydraulic Flow Diagram System 1

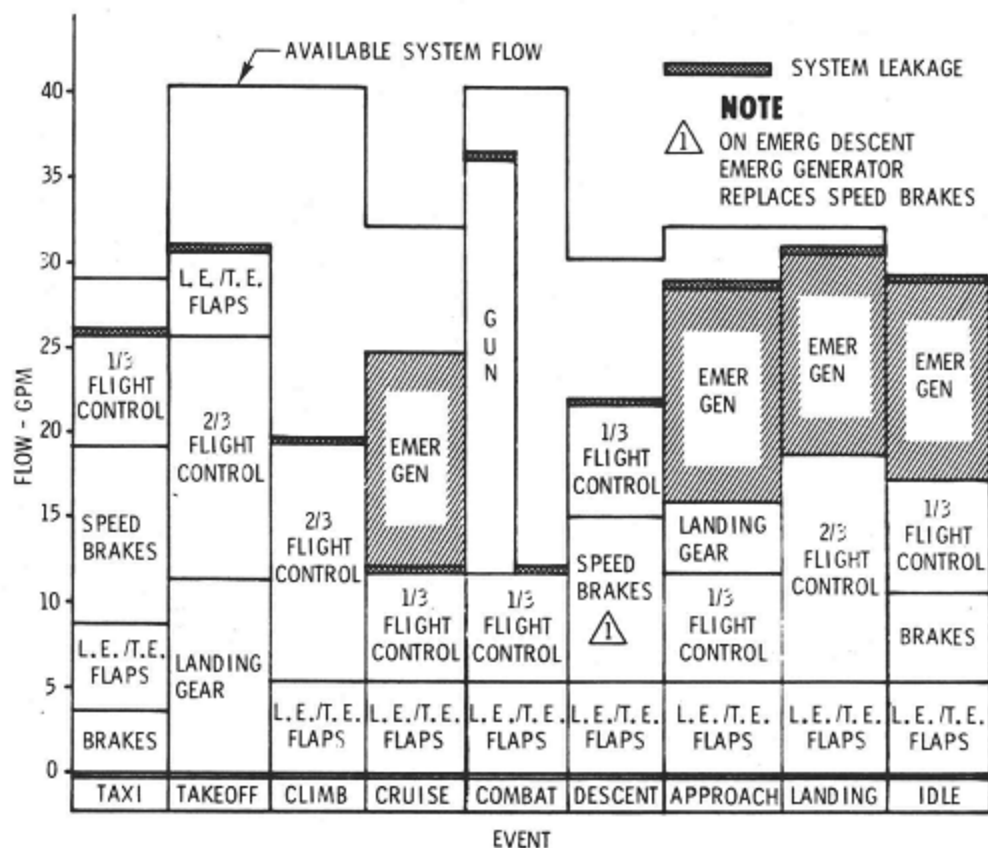


Figure 4-25. Hydraulic Flow Diagram System 2

rates to maintain flight until recovery of engine rpm or to enable an emergency landing to be made.

The landing gear has free-fall capability so that no emergency extension provisions are required. A hydraulic motor/generator is powered by the No. 2 system for the standby electrical power source. Brazed joints and fittings are used throughout the hydraulic system for low maintenance and high reliability. The systems and components are in accordance with the intent of military specifications and the system uses standard aircraft hydraulic fluid.

4.5 LANDING GEAR AND DECELERATING DEVICES

4.5.1 Landing Gear

The Lancer is equipped with tricycle-type, hydraulically-operated, forward-retracting landing gears. The main and nose gears are similar in design to those of the F-104S aircraft. The wheels, brakes and tires are designed to conform to the intent of applicable military specifications. The brakes use carbon heat sinks, are hydraulically powered, and include a modulated pressure-type anti-skid system. A hydraulic accumulator provides emergency power to the brakes. The main gear shock loads are primarily absorbed by a liquid spring. The retraction actuator locks mechanically in the extended position to react fore and aft loads. Each main gear has a one-piece, hydraulically-operated, electrically-sequenced door which, when closed, engages two uplocks to hold the gear in the fully retracted position.

The nose gear employs an air-oil strut to absorb shock loads and a linkage mechanism which locks mechanically to react the fore and aft loads when the gear is extended. The nose wheel is power steered through an angle of $\pm 25^{\circ}$, has a 3-inch trail, and is shimmy-damped through the steering actuator. The nose gear doors are mechanically actuated by the gear. When fully retracted, the gear engages an uplock which in turn holds the doors closed.

A manual system permits release of the gears in event of a hydraulic system malfunction. Gravity and air forces lower the gears to the fully extended position, and they are locked down by mechanical devices.

4.5.2 Drag Chute

The drag chute system is similar to that used on the F-104S. Deployment of the chute for normal and emergency stops will extend the brake, tire, and wheel life by



reducing brake operating temperatures. Normal and emergency stops may also be accomplished without the drag chute.

4.5.3 Speed Brakes

Clam shell type speed brakes are located in the lower fixed portion of the vertical tail, and are hydraulically operated. In the retracted position, they form the vertical tail contour. The speed brakes can be modulated by the pilot from 0 deg to 60 deg extension. They are designed to extend symmetrically and require minimum pitch trim correction when extended. The larger speed brake projected area of 11.9 square feet provides increased deceleration capability.

4.6 ELECTRICAL SYSTEM

4.6.1 AC Power

In the Lancer, the basic ac electrical power source is a 40/50-kva, 400 Hz, three-phase regulated-frequency generator mounted on an integrated constant speed drive powered by the engine. In addition to the main generator, a standby 10-kva hydraulically-driven generator provides fixed frequency (400-Hz) power. Emergency electrical power is provided by a 6.5-kva, 400-Hz generator mounted on and driven by the ram-air turbine. Protection, regulation and controls are provided for the various systems. A power profile is shown in Figure 4-26 and the ac schematic diagram is shown in Figure 4-27.

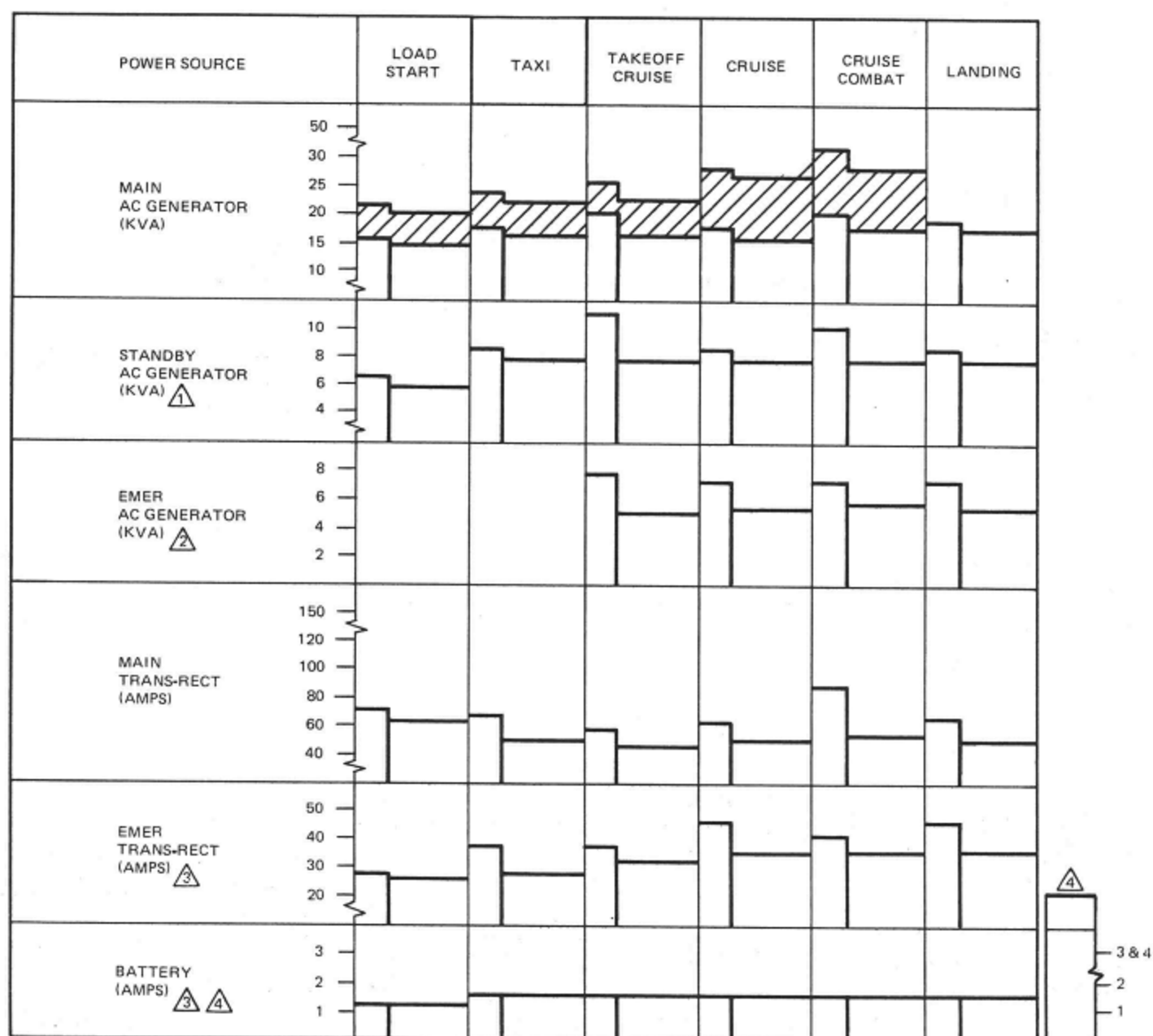
4.6.2 DC Power

The basic dc electrical power source is a 150-ampere transformer-rectifier (TR). A 50 ampere TR unit furnishes emergency dc power. A 24-volt, 13-ampere-hour battery provides for emergency electrical power and engine starting. Figure 4-28 presents a schematic diagram of the dc power generation system.

4.7 ENVIRONMENTAL CONTROL SYSTEM

The Lancer environmental control system has been optimized for the performance requirements and provides adequate cooling capacity for additional optional electronic equipment.

A simple turbine-fan air cycle system uses high pressure and high temperature engine bleed air, with ram air and engine fuel as heat sinks. The equipment installation is completely accessible from the ground through an external door in the right side of the fuselage. Primary functional and safety subsystems (such as



NOTE: ^① LEADS SHOWN WITH MAIN AC GENERATOR INOPERATIVE
^② MAIN AND STANDBY GENERATORS INOPERATIVE
^③ MAIN TRANS - RECT INOPERATIVE
^④ ENGINE START 5 SEC - 329 AMPS
 15 MIN - 2.1 AMPS


 CALCULATOR
 BASIC LOADS


 CALCULATED

0 5 SEC
 15 MIN
 LOAD PROFILE
 SCHEDULE

Figure 4-26. Electrical Power Load

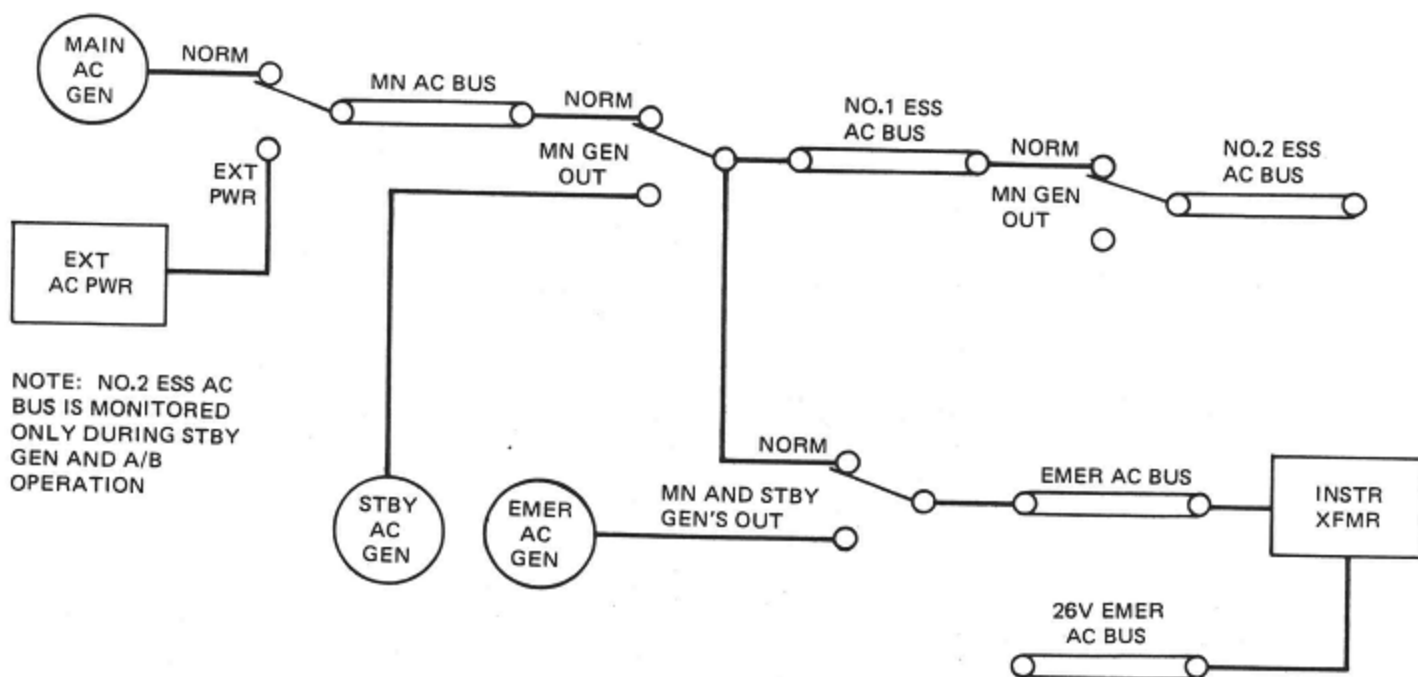


Figure 4-27. AC Bus System

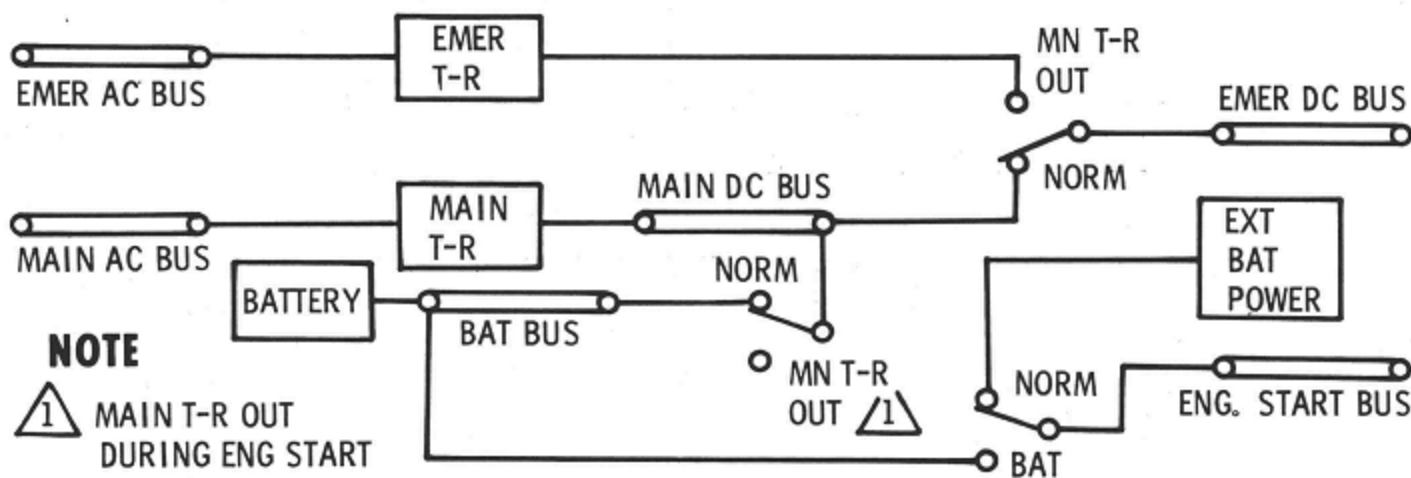


Figure 4-28. DC Bus System

windshield and canopy defogging, radar pressurization, and gun purge ejector) are branched off upstream of the refrigeration unit to enable these systems to function even though the refrigeration unit is shut off. The schematic diagram presented in Figure 4-29 shows a complete system definition.

The air conditioning system can provide cockpit supply temperatures of 35°F to 100°F during all flight and ambient conditions, and can maintain a comfortable supply temperature of 80°F during a hot day, at engine idle rpm condition on the ground.

In order to maintain efficiency during the stress of combat, the pilot may adjust the flow and direction of cockpit-conditioned air over the upper portion of his body. Fixed outlets supply air to the pilot's feet and lower body.

4.8 LIGHTING

Lighting for the Lancer includes landing, taxi, navigation and formation lights for exterior lighting and instrument panels, console panels, flood, spot and thunderstorm lights for interior lighting. Warning and caution lights are also used in the cockpit for protection and safety.

4.8.1 Exterior Lights

One landing light is installed on each main landing gear and a taxi light is provided on the nose gear. Anti-collision lights are installed on the top and bottom of the fuselage. Wing tip and tail lights are provided. The missile launcher, which covers the wing tip light when installed, incorporates a formation light. The optional tip tanks also include formation lights.

4.8.2 Interior Lighting

White, integral lighting adequate for all-weather and day and night flight operation is provided in the Lancer instruments and consoles. Two incandescent, unfiltered thunderstorm lights are installed in the cockpit. Warning and caution lights are placed on full bright automatically when the thunderstorm lights are operated, and when the instrument light control is turned off (bright daylight). A removable incandescent unfiltered map light with coiled cord is located on the right-hand console.

4.8.3 Warning and Caution Lights

An illuminated knob is provided on the landing gear control lever for ease of identification, and landing gear "down and locked" lights are also provided. In



addition, a landing gear warning light is installed on the left side of the main instrument panel.

A composite warning panel is installed to warn the pilot of unsafe or abnormal conditions. The master caution light will illuminate whenever one of the warning lights on the composite warning panel is energized. Warning lights are also provided elsewhere on the instrument panel for engine air inlet temperature, master caution, fire, canopy unlocked, landing gear unsafe, and slow flight.

4.9 AVIONICS SYSTEM

The Lancer inherits from the F-104S a large avionics bay with adequate power, wiring and cooling for a complete multipurpose avionics suite comprised of the older vacuum tube type components. The Lancer could be produced and operated with these same systems if an Air Force so chooses. The procurement and logistics advantages to an Air Force planning to retain and operate a substantial number of F-104G or F-104S aircraft with the Lancer might justify such a decision. Lockheed, however, does not recommend this alternative for the production Lancer configuration, for several reasons.

The Lancer must be tested in both the laboratory and in flight through complete weapon system qualification. Therefore, a minimum, non-recurring development and test cost for a new avionics suite is realized by combining the avionics testing with the required aerodynamic, structural and system testing. Much of this non-recurring development testing will be accomplished in the Lockheed/Aeritalia first phase program. In all instances, there is available today a modern, solid state item of avionics equipment that is already developed and in service and that is at least the functional equivalent of the system installed in the F-104S. Lockheed has examined all of these systems in detail to identify the potential effectiveness and reliability improvements balanced against the procurement, systems integration and logistics costs. In each case studied, there is a life cycle cost saving in favor of investing in newer "off-the-shelf" equipment. These savings occur because of the increased reliability and built-in test features available in the newer equipment. There is also an implied improvement in operational readiness, utilization, maintenance manpower, and average turn-around time that could be exploited to diminish operating costs, or fly more hours at a fixed operating cost. An additional important consideration is the increased probability of mission success with higher MTBF equipment.

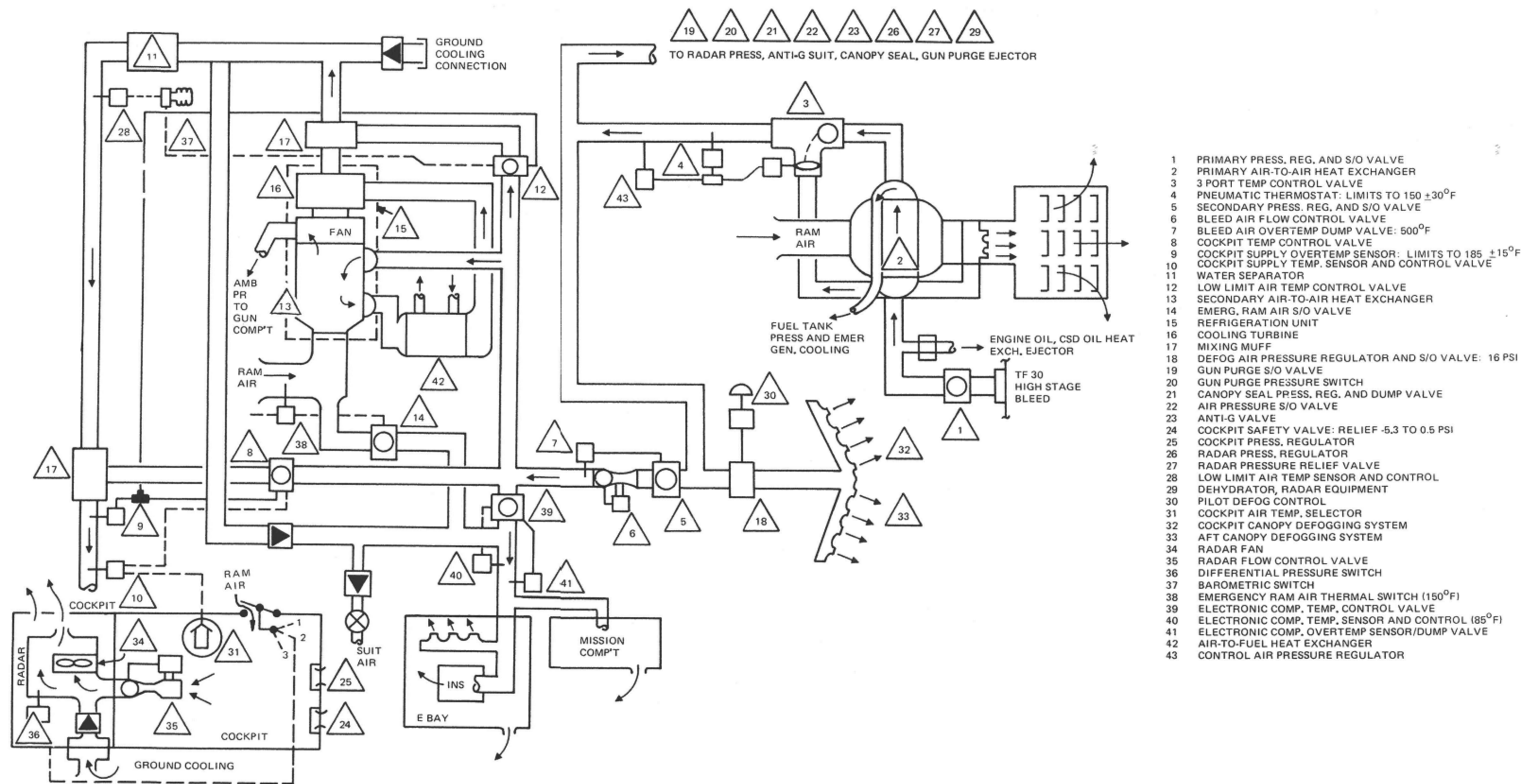


Figure 4-29. Schematic ECS

It must be emphasized that the cost savings discussed in the preceding paragraph are only attained by selecting a newer, service-proven system and doing the integration and qualification testing concurrently with the Lancer development testing. In no case studied could a clear cost savings be proven for a new system that had to be developed from the early development or "breadboard" phase, into a qualified and operational system. An advantage to using existing systems in the Lancer that cannot be isolated in any cost analysis is the existence of depot maintenance and overhaul facilities, spare parts availability, and larger quantity procurement benefits.

In this avionics section, the equipment that Lockheed recommends for a multimission Lancer is described in detail. Lockheed recognizes that the ultimate selection of equipment will be made by the using Air Forces on the basis of mission and cost effectiveness priorities. The flexibility of the Lancer avionics bay to accommodate almost any combination of the newer systems provides assurance that any Air Force's requirements can be met with the available volume, cooling and power. The avionics equipment described in this section is typical of modern solid state systems. There are other combinations that will be available and could be substituted in the Lancer aircraft.

One particularly troublesome area in selecting an avionics system deserves explanation because of its contribution to both mission effectiveness and weapon system cost. The radar systems developed since the R21G/H in the F-104S do not provide the functional capabilities of the R21G/H for anything approaching its cost, size, weight or power. The new air-to-air, range-only equipment in the F-5E has no capability for the Sparrow missile or a ground mapping mode. The MRCA air-to-ground attack system has no intercept function and is only marginal in any air-to-air mode. At the other end of the spectrum, there are the extremely sophisticated, expensive, large and heavy systems for the F-14 and F-15. There appear, therefore, to be only two alternatives to continuing to use the R-21G/H. In Lockheed's recommended configuration, the assumption has been made that the new Westinghouse WX100 radar or WX200 will be funded for some other application and will, therefore, be available for adaptation to the Lancer.

In the event this is not the case, Lockheed will investigate other radar manufacturers using current modular solid state technology to achieve a functional, but more reliable equivalent of the R21G/H, or the design of the R21G/H will be investigated to modernize it using current solid state technologies.



Table 4-6 compares the current list of F104S avionics with alternates and gives Lockheed's recommendation. Specifically, the Lancer avionics design described here incorporates modular units with minimum interfaces, thus increasing mission reliability while reducing maintenance time. This modular design also allows relatively simple replacement as newer equipment designs become available. A complete block diagram for the avionic systems is presented in Figure 4-30. Significant characteristics of systems that could replace the F-104S existing systems are listed on Table 4-7. Other similar equipment from U.S. or foreign sources is also available for installation in the Lancer.

In the Lancer, avionics are installed in three areas: the electronics bay just aft of the cockpit, the cockpit, and the nose (see Figure 4-2). As shown in Figure 4-31, all electronics bay equipment is mounted on a two-shelf rack that can be raised clear of the fuselage on a center post by means of compressed air or from a mechanically operated system, thus allowing unrestricted access. Electrical and cooling functions remain operative in the raised position. Access to cockpit equipment is provided both through the cockpit and through removable panels in the side of the fuselage. The nose is attached with four latches and is easily removed for access to the radar equipment.

The currently recommended fire control, communications, identification, navigation and electronic warfare equipment are described in the following paragraphs.

4.9.1 Fire Control System

The fire control system currently recommended for the Lancer includes the radar unit, display unit, symbol generator, and the lead computing gyro unit.

For the radar, Lockheed recommends a high power, angle tracking system for air-to-air gunnery, missile attack, ground bombing and mapping functions. For a medium range unit, equipment such as the new Westinghouse WX-100 or Texas Instrument TIR 2100 radars, containing solid state line replaceable units with self-test features can be installed to track 5 square meter aerial targets to 20 nautical miles (nm). The modular construction of the WX-100 permits use of components and expansion into higher performance units such as the WX-200, for tracking targets through ground clutter and ranges to 35 nm, depending on antenna size and maintenance losses.

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TABLE 4-6. ALTERNATE LANCER AVIONICS (Continued)

FUNCTION	F-104S	LANCER		LOCKHEED RECOMMENDATION
		ALTERNATE 1	ALTERNATE 2	
<u>TACAN</u> Provides distance and bearing to any selected TACAN beacon line of sight Status	AN/ARN-52 Foreign Prod. AN/ARC-552	RT-1127/ARN-84(V) F111B and T38 AN/ARC-150V2	AN/ARN-91 F4K, Harrier AN/ARN-109	RT-1127/ARN-84(V) AN/ARC-150V2
<u>UHF Communication</u> Provides two-way air-to-air and air-to-ground voice communication plus a separate guard channel receiver. Status	Foreign Prod.	F5E	F111, C5A Prod.	
<u>Emergency UHF</u> Provides two-way air-to-air and air-to-ground voice communication in the event of failure of pri- mary UHF. Status	TR3 Foreign Prod.	AN/ARC-150V2	---	AN/ARC-150V2
<u>Intercom</u> Provides amplification of transmitted and received radio signals and communication between cockpit and ground crew. Status	Foreign Prod. AN/AIC-18 (part of UHF)	F5E C6533/ARC Provides indivi- dual selection of transceivers and warning tones Light Obs. Helicopter	AN/AIC-18 Same as C6533/ARC SR-71	C6533/ARC
<u>IFF</u> Enables aircraft to identify itself from interrogating systems. Status	AN/APX-46 AIMS Mark X Capability Foreign Prod.	AN/APX-72 AIMS Mark XII Capability F4, F5E, F14	AN/APX-101 AIMS Mark XII Capability with transponder diversity F15	AN/APX-72

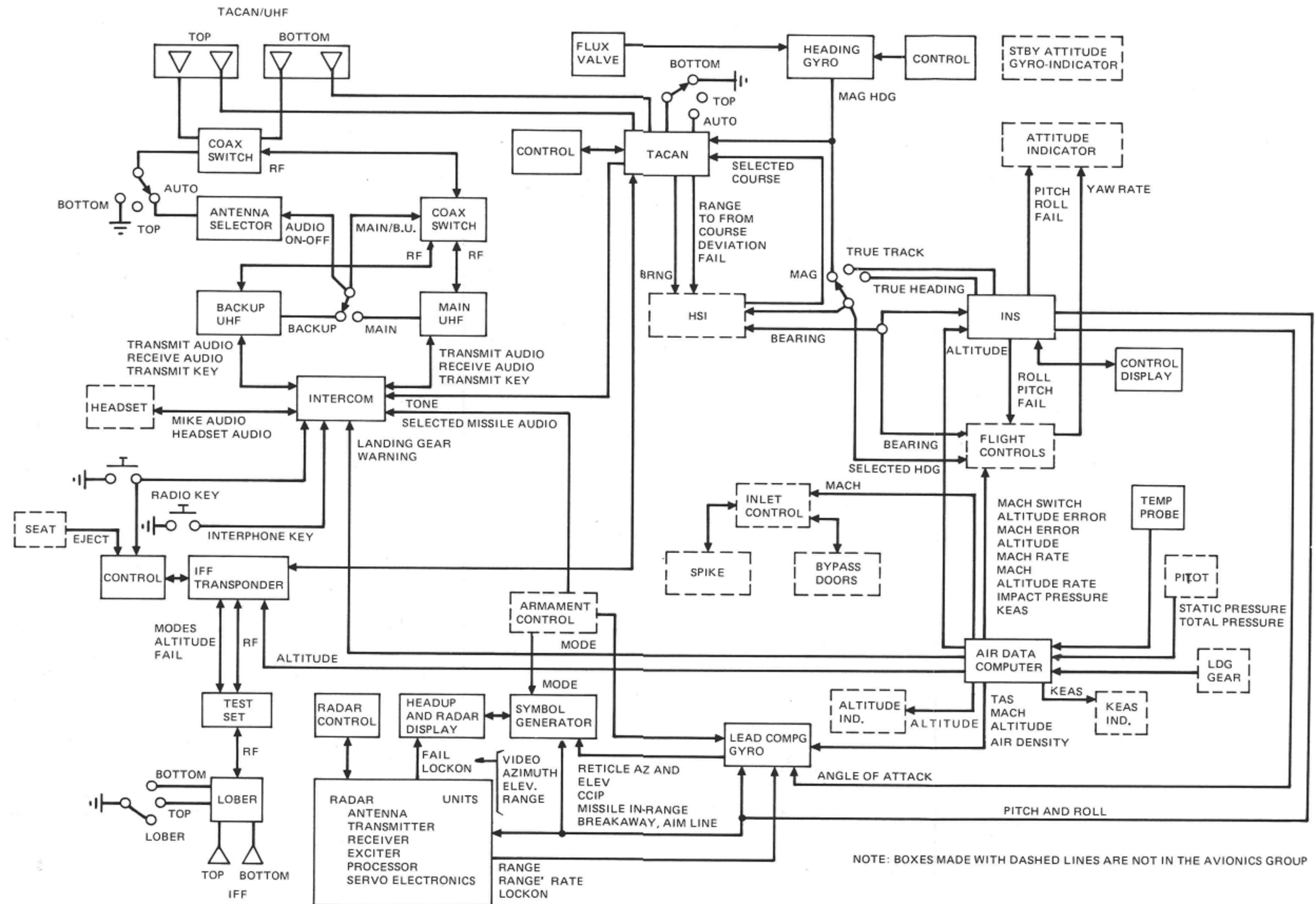


Figure 4-30. Avionics Block Diagram

TABLE 4-7. POSSIBLE LANCER AVIONICS - LEADING PARTICULARS

SYSTEM	CHARACTERISTICS	SYSTEM COMPONENTS	MFG.	WEIGHT LBS.	VOLUME CU. IN.	POWER 115 VAC 400 HZ	WATTS 28 VDC	COOLING AIR LBS./MIN.
Radar WX-200	Search and track 37 nm range on 5m ² target - 9.6 - 10.25 GHz Coherent pulse doppler Track through clutter wide angle scan	Rcvr-Xmtr antenna	Westinghouse	350	10,000	5,000		10.0
Display (HUD)	CRT HUD with reticle and aim line DVST for radar gun boresight cross nav and flight symbols Capable for TV format HUD visible against 10,000 foot lambert background	Display Symbol generator	G.E.	42	1,400	430		---
Lead Computing Gyro AGS-6	Snapshoot or tracking on maneuvering targets Air-to-ground CCIP Launch envelope for several missiles	Gyro-computer	G.E.	19	900	230	5	---
INS LN-30	2 minute fast align 1.5 nm/hr CEP 8,000 word X24 bit memory 50% excess capacity	Inertial unit Display	Litton	43	1,880	360	-	1.0
Gyro Compass AN/ASN-89V	Automatic synchronization No controller required +10 slaved mode +5.59/HR free gyro mode	Gyro Flux valve Compensator	Sperry	9	360	40	-	---
TACAN RT-1127/ARN-84(V)	All modes (X, Y, air-to-ground, air-to-air ranging) No false bearing lock	Control RCVR-XMTR Coax switch	Hoffman	38	1,260	345	5	---

TABLE 4-7. POSSIBLE LANCER AVIONICS - LEADING PARTICULARS (Continued)

SYSTEM	CHARACTERISTICS	SYSTEM COMPONENTS	MFG.	WEIGHT LBS.	VOLUME CU. IN.	POWER 115 VAC 400 HZ	WATTS 28 VDC	COOLING AIR LBS./MIN.
UHF AN/ARC-150V2 (primary)	Automatic antenna Switching 300 NM range 10 Bearing Accuracy Range within 0.04% All solid state compatible with ADF and Secure speech keyers 3 V Rcvr sensitivity 7,000 channels 10 watt output 20 preset Ch. plus guard	Antenna (2) (part of UHF) Control Rcvr-Xmtr Coax Switch Antenna (2) Antenna select	Magnavox	20	440	---	103	---
UHF Backup AN/ARC-150V2	Same as primary	Control Rcvr-Xmtr Coax switch	Magnavox	14	410	---	98	---
Interphone C6533/ARC	Monitors and receivers or audio inputs Control to 5 transmitters Interphone line	Control	Andrea	2	80	---	4	---
IFF AN/APX-72	500 Watt Output All attitude antenna coverage FAA traffic control Complies AIMS MK XII	Transponder Control Test set Lobster Antenna (2)	Honeywell	25	740	99	27	---
ADC	Analog AIMS altitude reporting	Computer	Airesearch	18	660	330	--	---
TOTAL AVIONICS				580	18,130	6,834	242	11.0

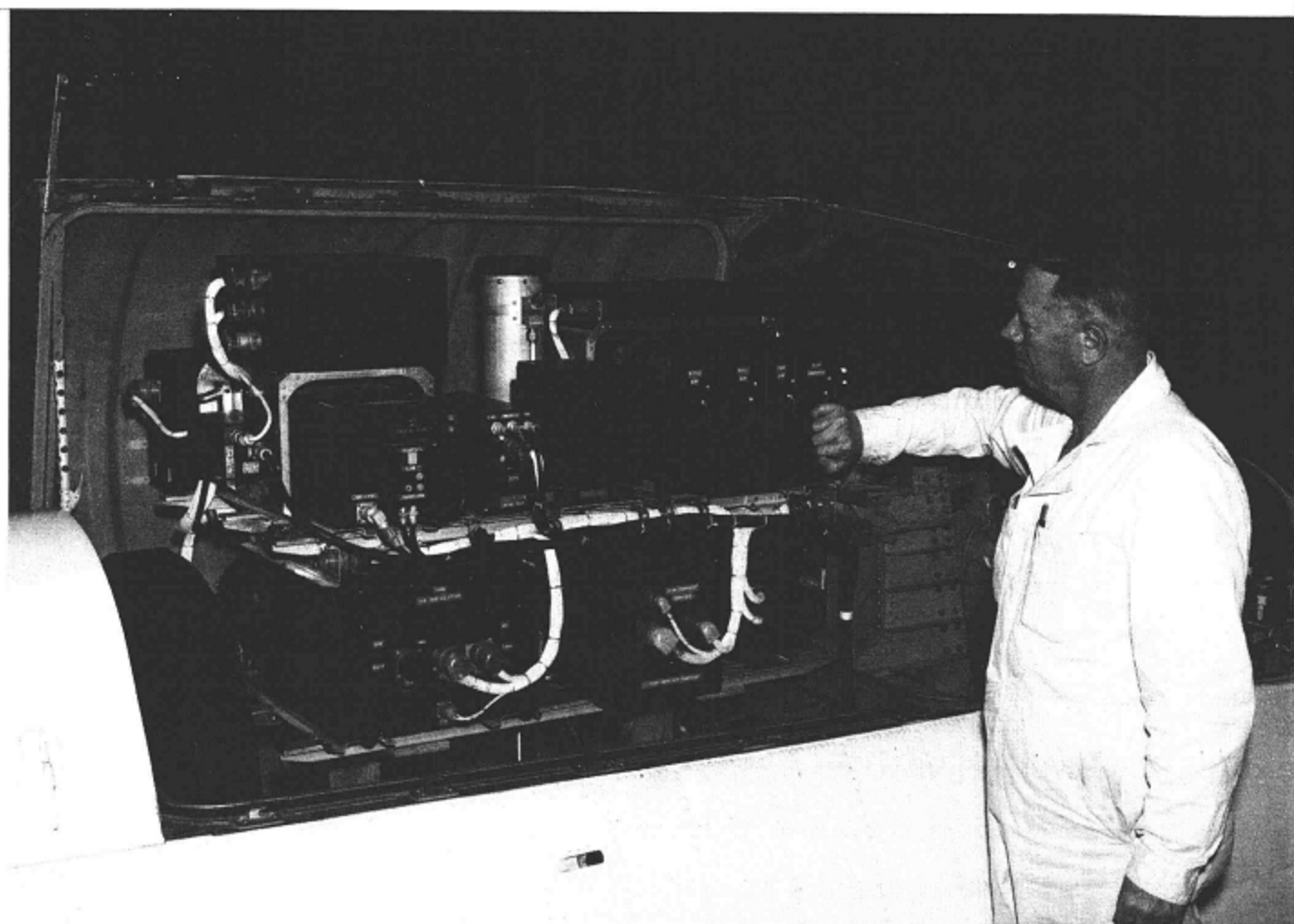


Figure 4-31. Avionics Rack

A multi-purpose display unit is proposed for presentation of radar data and computer symbology for weapon delivery by repackaging proven components. The radar presentation consisting of a Direct View Storage Tube (DVST) is on the front face of the unit. Behind the DVST for weapon delivery functions is a 2.5-inch Cathode Ray Tube (CRT) with optics which provide a 4-inch aperture reflected onto a combining glass and collimated so as to appear at infinity. A 20 degree field of view is available, from 5 degrees above to 15 degrees below the optical centerline.

The display unit is compatible for presentation on the combining glass of television format as currently used for a number of air-to-ground weapons. To provide sufficient contrast in high ambient light conditions, the combining glass requires some degree of opacity, easily accomplished by elevating an opaque flag to obscure the center section when selecting a TV mode.

The lead computing gyro unit (AGS-6) was developed during the late 1960's for aerial combat against maneuvering targets. It generates rate and acceleration signals from aircraft motion. It uses range/range rate data from the radar and angle of attack, true air speed, and altitude data from the Air Data Computer. These data are used for correct positioning of an aiming dot or line presentation on the Headup Display (HUD), for firing against either maneuvering or non-maneuvering targets. Computations may be stored in the computer unit for the ballistics of numerous missile systems. The configuration of the lead computing gyro unit being utilized in a disturbed reticle or line type aiming reference easily allows for future replacement with "strap down" gyro/digital computer units.

4.9.2 Communication and Identification Equipment

Two 10-watt UHF/AM radios are installed in the Lancer for voice communication, each containing 7000 selectable channels. Twenty channels plus guard may be preset. An intercommunication control unit is provided for selection of up to five transmitters and six switches for the individual monitoring of audio receiver inputs. These include TACAN, IFF, Landing Gear Warning and Missile Lock-on.

The currently proposed identification transponder (AN/APX-72) is in volume production and meets all U.S. Tri-Service and FAA requirements of the AIMS MK XII program.

4.9.3 Navigation Equipment

Navigation is provided by inertial, TACAN, and Backup Heading Gyro systems.



The inertial navigation unit is a self-contained lightweight system providing an accuracy of 1.5 nm/hour CEP. The inertial measuring unit contains the platform with two gyros and three accelerometers, a digital computer, a converter card, and platform electronics with a power supply. The computer is general purpose, 24-bit with 8000-word capacity and a 2-MHz clock frequency.

A solid state microtacan set is installed using the basic units of the AN/ARN-84 and AN/ARN-91 (ANS 900 Series) receiver transmitter units used in many U.S. military and foreign aircraft. The TACAN system antennas (located on top and bottom fuselage) are dual UHF/TACAN units with individual and automatic control for each system. Backup heading data are provided by the AN/ASN-89V unit and flux valve and are used as a magnetic reference for the TACAN system.

4.9.4 Electronic Warfare Equipment

Installation of electronic warfare (EW) equipment, if required, would be modularized to fit in the 30-inch fuselage section aft of the ammunition bay with some reduction of internal fuel capacity. Modular concept permits alternate EW equipment configurations to match the needs of different missions. In the interceptor or air superiority role, protection could include active jamming against surface-to-air missile systems (SAM) and warning against tail attack by airborne fire control radar (AI), as follows:

<u>Threat</u>	<u>Approximate Box Size</u>	<u>Power Required</u>
SAM	4 ft ³	5 KW
AI	0.5 ft ³	0.1 KW

For the interdiction or strike fighter role, protection could include active jamming against radar directed AAA fire and warning against SAM or AI attack, as follows:

<u>Threat</u>	<u>Approximate Box Size</u>	<u>Power Required</u>
AAA	2 ft ³	2.5 KW
AI	0.5 ft ³	0.1 KW
SAM	1.0 ft ³	0.25 KW

Antennas for AI and SAM warning would be permanently installed at various locations on the aircraft, but antennas for the active jammers could be part of the respective modular package.

4.10 OXYGEN SYSTEM

A service proven, 5-liter liquid oxygen (LOX) system is provided in the Lancer. The system consists of a LOX converter, heat exchanger, and control and display panel. This system provides the pilot with approximately four hours of breathing oxygen. For emergency, a 100-cubic-inch gas bottle, mounted in the Stencel seat, provides oxygen for 15 minutes, either automatically upon bailout or manually by means of a pull cord, called the "green apple," located at the pilot's side.

4.11 ARMAMENT SYSTEM AND EXTERNAL STORES

The proposed basic Lancer armament system consists of a fixed internal gun installation and seven external store positions on wing and fuselage pylons.

4.11.1 Internal Armament

The proposed gun installation consists of one General Electric M-61 20-mm hydraulically-driven gun and a 500-round single-end linkless ammunition feed system, both of which will reflect the very latest design features available at time of aircraft manufacture. The normal firing rate will be 6000 rounds per minute. These are proven components adapted from other USAF Fighters. An alternate 3-barrel version of this gun is possible to save weight.

The gun compartment is located in the lower left half of the forward fuselage section, with access for service through two compartment doors. Spent cases are retained in a compartment in the lower half of the forward fuselage. A purge system is provided to remove gun gases from both the gun and spent case compartments. This installation is similar to that in the F-104S.

Other alternate gun installations can be provided if desired. For instance, the forebody can be adapted for a dual 30-mm D.E.F.A. or ADEN gun installation with 200 rounds of ammunition per gun in lieu of the M-61 20-mm gun with its 500 rounds of ammunition. This alternate installation requires one smaller gun compartment on each side of the forward lower fuselage. The ammunition will be stowed in two lightweight containers installed in the same area as the M-61 ammunition linkless feed system drum. Both cases and links will be retained and the purge system will be similar to that provided with the M-61 installation.



4.11.2 External Armament

Three store stations are provided on each wing, positioned at W.S. 175 (tip), at W.S. 152, and at W.S. 112, and one on the fuselage centerline. The weapon and store carrying capability chart (Figure 4-32) shows various types of external stores that can be carried from individual stations as well as the type of ejector rack used. The pylons can be adapted to carry advanced, future stores.

4.12 CREW STATION

The Lancer crew station is identical in external contours and dimensions to that of the F-104S. It is designed to provide 360 degrees external vision in the upper hemisphere, except for minimum local blanking by the windshield brow ring, thin canopy frames and seat headrest. This blanking can be reduced by utilizing body and head motion permitted by the cockpit dimensions. The Lancer utilizes a simplified instrument panel design with a thin glareshield and no instruments protruding into the external vision area. Rear vision is augmented by two internal, canopy-mounted mirrors.

The cockpit is sized to accommodate the 5 to 95 percentile pilots in an ejection seat with zero-zero (speed-altitude) capability. The canopy height is minimized for best aerodynamic performance while providing for adequate target/gun lead angles, and vision over the nose. A full five inches of vertical adjustment of the seat bucket is provided. The side consoles are sloped 15 degrees to provide adequate vision and reach of all system control functions.

Pilot work load is minimized during combat by grouping critical flight control, armament, navigation, and engine functions on the forward instrument panel in a manner to provide good vision and easy reach (see Figure 4-33). A unique feature provides for release of external weapons through the "Pickle Button" on the B8A stick grip and the separate control of gunfire through the trigger. This allows the pilot to transfer quickly from missile firing to air-to-air gunnery without manipulating additional weapon selection controls, permitting maximum headup and forward attention by the pilot. In addition to the normal visual mode for Sidewinder delivery and gunfire, the combined headup and radar display could provide a radar positioned target symbol on the headup display (HUD) to assist the pilot in tracking intermittent visual targets.

NOTE

1 THIS IS NOT A MISSION LOADING CHART. ANY CROSS-MIXING OF STORES SHALL BE DONE WITH UTMOST DISCRETION TO PRECLUDE HARDWARE INTERFERENCES AND/OR ADVERSE STORE RELEASE AND SEPARATION CONDITIONS.

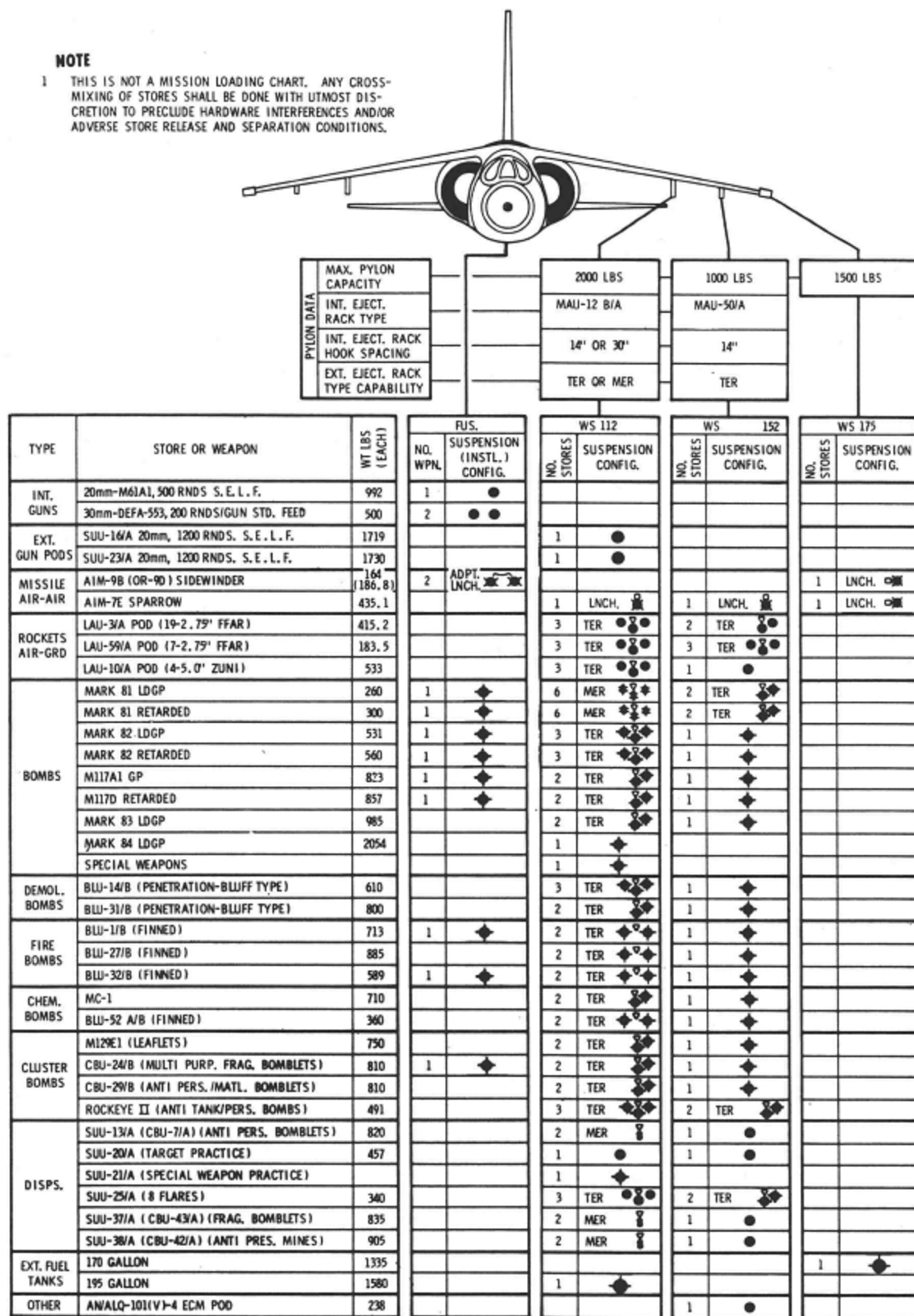
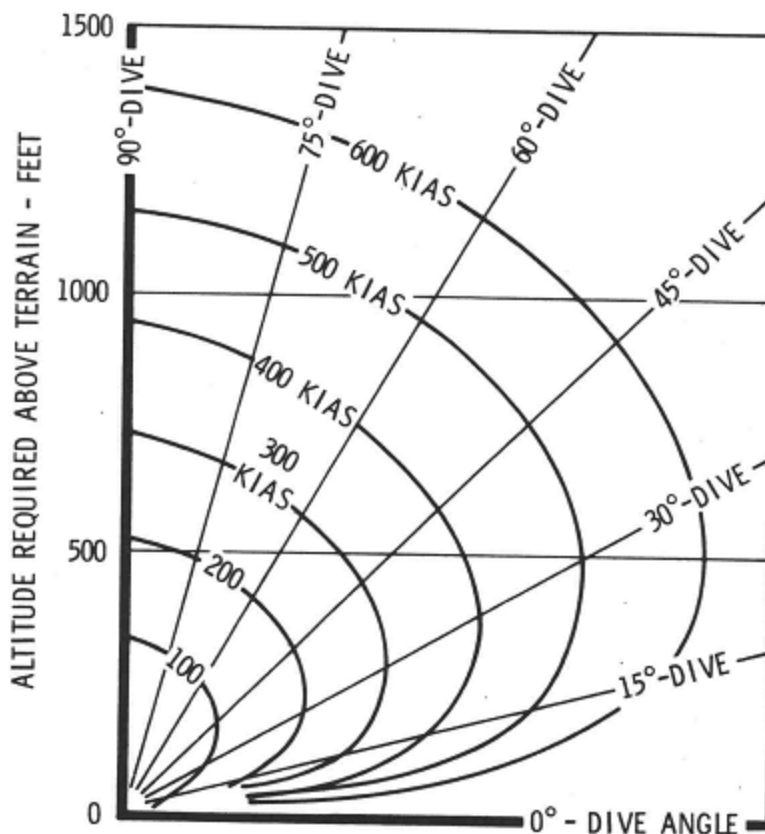


Figure 4-32 Stores

As shown in Figure 4-33, the main instrument panel is in three parts: attitude and engine instruments are on the right side; air data and temperature instruments are on the left side; the large center section can accommodate several variations in combined radar (HUD) displays or optical sights, without requiring complete revision of the other sections of the main instrument panel. The lower instrument panel provides additional versatility in that a total of eleven inches of standard control panels may be installed in the center stand or interchanged with control panels in the side consoles without structural modifications.

4.13 ESCAPE SYSTEM

The basic configuration is designed to accommodate the Martin-Baker crew escape system. A comparative analysis of all available crew escape systems showed that the newly developed Stencel Model S-4 ejection seat would provide greater capability at a lower weight. The adoption of the S-4 seat is provided as an option. The S-4 seat provides escape capability from zero speed and altitude through 600 knots as shown in Figure 4-34. It incorporates seat stabilization in launch and high altitude descent, back-up ballistic systems to improve reliability, and a ballistically-deployed personal parachute system to augment low-level escape.



MINIMUM EJECTION ALTITUDE
REQUIRED AS A FUNCTION OF
VELOCITY AND DIVE ANGLE

NOTE

CURVES ARE FOR ZERO
PILOT REACTION TIME,
AND BASED ON WINGS
LEVEL BANK ATTITUDE

Figure 4-34. Ejection Seat Envelope

